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USAAVLABS TECHNICAL REPORT 66-45

XV-4A VTOL RESEARCH AIRCRAFT PROGRAM

Summary Report

By

Robert Nicholson

Randall B. Lowry

May 1966

U. S. ARMY AVIATION MATERIEL LABORATORIES

FORT EUSTIS, VIRGINIA

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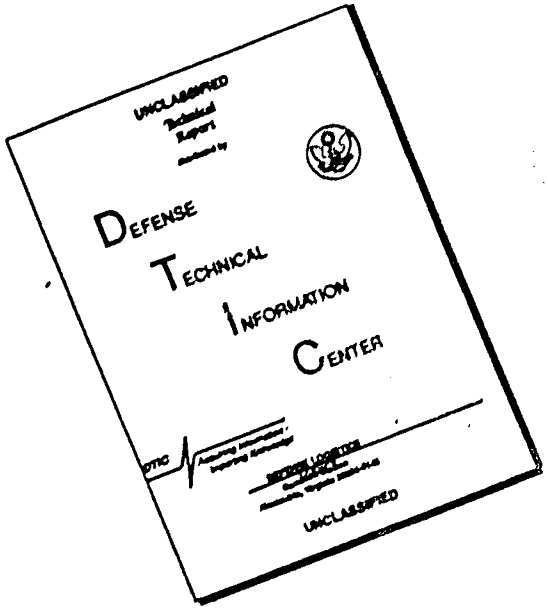

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ABSTRACT

The results of the XV-4A vertical takeoff and landing (VTOL) research aircraft program, including a review of the aircraft design, aircraft systems, flight test program, VTOL lift improvement program, and small-scale and full-scale wind tunnel tests, are presented in this report.

FOREWORD

This report summarizes the work performed during the XV-4A VTOL research aircraft program. The primary purpose of the program was to determine the feasibility of the augmented jet ejector concept for attaining a VTOL capability for aircraft. The program was initiated on 30 June 1961 and was concluded on 30 September 1965.

The work was performed by the Lockheed-Georgia Company, Marietta, Georgia, under Contracts DA 44-177-TC-773 and DA 44-177-AMC-14(T) with the U. S. Army Aviation Materiel Laboratories. Mr. E. B. Gibson, Chief Advanced Design Engineer, and Mr. A. W. Mooney, Project Engineer, directed the program at Lockheed.

This summary report was prepared at the U. S. Army Aviation Materiel Laboratories under House Task AA 65-35.

CONTENTS

ILLUSTRATIONS

SYMBOLS

A/S	airspeed
BLC	boundary layer control
CG	center of gravity
CTOL	conventional takeoff and landing
EPR	engine pressure ratio
f	ejector resultant force displacement, inches
F	thrust, pounds
F_a	control stick lateral force, pounds
FAT	free air temperature
F_e	control stick longitudinal force, pounds
FMSS	flight mode selector switch
F_r	rudder pedal force, pounds
FS	fuselage station
GR WT	gross weight
H_{ic}	altitude, instrument corrected
I_x	moment of inertia about the X axis, slug-foot^2
I_y	moment of inertia about the Y axis, slug-foot^2
I_z	moment of inertia about the Z axis, slug-foot^2
KEAS	knots equivalent airspeed

KIAS	knots indicated airspeed
MAC	mean aerodynamic chord
N. A.	not available
n_x	longitudinal acceleration, g
n_y	lateral acceleration, g
n_z	vertical acceleration, g
Phase I Flight	both engines in lifting mode; speed range from 0 to approximately 80 knots
Phase II Flight	left engine in thrusting mode and right engine in lifting mode; speed range from approximately 80 knots to approximately 120 knots
Phase III Flight	both engines thrusting with ejector doors open; speed range from approximately 120 knots to a maximum of 170 knots
P_t	total pressure
SAS	stability augmentation system
T_t	total temperature
V_{ic}	airspeed, instrument corrected
VTOL	vertical takeoff and landing
W_B	distance between trailing edges of ejector inboard walls, inches
WL	water line
α	angle of attack, degrees
β	angle of sideslip and ejector exit outer door angle, degrees
δ_a	aileron angular displacement, degrees

δ_{AMB}	ratio of ambient pressure to sea level standard day pressure, $\frac{P_{AMB}}{PSL}$
δ_e	elevator angular displacement, degrees
δ_r	rudder angular displacement, degrees
δ_{stick}	control stick longitudinal angular displacement, degrees
δ_{stick_a}	control stick lateral angular displacement, degrees
θ	pitch angle or ejector resultant force inclination, degrees
$\dot{\theta}$	pitch rate, degrees per second
ϕ	bank angle, degrees
ϕ_n	ejector system augmentation ratio
$\dot{\phi}$	roll rate, degrees per second
$\dot{\psi}$	yaw rate, degrees per second

SUMMARY

A program has been conducted to determine the feasibility of the augmented jet ejector concept for attaining a VTOL capability for aircraft. During the flight test program, the actual vertical thrust realized was only about 93 percent of that predicted, and consequently the aircraft, the XV-4A, had a marginal lift capability. This marginal lift capability severely limited the capability to conduct quantitative data gathering during the flight test program. This report presents the limited quantitative results obtained and a brief summary of the aircraft design, systems, flight test program, VTOL lift improvement program, and small-scale and full-scale wind tunnel programs.

The feasibility of the augmented jet ejector concept has been demonstrated; however, this concept is not considered to be competitive with other concepts for attaining a VTOL capability.

CONCLUSIONS

It is concluded that:

1. The feasibility of the augmented jet ejector concept for attaining a VTOL capability for aircraft has been demonstrated. However, only about 93 percent of the predicted aircraft lift capability was realized during the flight tests, and the ejector concept is not considered to be competitive with other VTOL concepts.

2. Reaction control systems utilizing turbojet engine compressor and exhaust bleed gases are feasible and provide an excellent means of aircraft control during VTOL operations.

3. A rate-only stability augmentation system (SAS) is adequate for VTOL operations.

INTRODUCTION

During the past several years, much aeronautical research has been devoted to investigating various concepts for attaining a VTOL capability for high-speed, fixed-wing aircraft. Since this type of aircraft requires considerably more thrust for VTOL operations than for cruising flight at high subsonic speeds, a severe mismatch in the propulsion system requirements exists. Various means have been suggested whereby the aircraft propulsion system is designed for cruising flight and its thrust is then augmented during VTOL operations by some device. Such a device, the augmented jet ejector, was the basis for the VTOL propulsion system of the XV-4A research aircraft.

The XV-4A VTOL research aircraft program was established in June 1961 for the primary purpose of determining the feasibility of the augmented jet ejector concept for application to VTOL aircraft. Secondary purposes included the investigation of handling qualities requirements for VTOL aircraft and a semioperational evaluation of this particular aircraft. Due to technical problems encountered during the program, not all of the objectives were attained. The program was concluded in September 1965.

The XV-4A aircraft, shown in hovering flight in Figure 1, was formerly designated the VZ-10 and is also known as the Lockheed Hummingbird, Model L-330.

This report contains a description of the XV-4A aircraft and highlights the aircraft design characteristics, aircraft systems, flight test program, VTOL lift improvement program, and wind tunnel tests. Some of the material in this report has been adapted from reports submitted by the Lockheed-Georgia Company during the XV-4A program; however, portions of the material presented here have not previously been published.

AIRCRAFT DESIGN AND GENERAL DESCRIPTION

The XV-4A, a twin-engine, midwing monoplane, is equipped with a propulsion system that permits vertical takeoff and landing (VTOL) and conventional takeoff and landing (CTOL) flight. Exhaust gases from the two Pratt and Whitney JT-12 turbojet engines flow rearward through conventional exhaust nozzles during CTOL flight and are directed into the augmented jet ejector system by diverter valves and then out the bottom of the fuselage during VTOL flight. The aircraft has a retractable tricycle landing gear and a T-type empennage. The general arrangement of the aircraft is shown in Figure 2, and the inboard profile is shown in Figure 3.

AIRCRAFT STRUCTURAL DESIGN AND DESCRIPTION

DESIGN

The XV-4A was structurally designed to meet the requirements of Part 3 of the Civil Air Regulations as applied to acrobatic aircraft. The structure was designed for a positive limit load factor of 6, a negative limit load factor of 3, a positive ultimate load factor of 7.5, and a negative ultimate load factor of 3.75. Thus, the overall factor of safety is not less than 1.25. However, in areas where a structural failure would unquestionably result in loss of the aircraft, the factor of safety is never less than 1.5 and is generally greater than 2.0. The structure was designed to withstand gusts of 66 feet per second at the maximum design cruise speed of 350 knots at sea level and gusts of 50 feet per second at the maximum design dive speed of 450 knots at sea level. The XV-4A design V-n diagram is shown in Figure 4.

The landing gear was designed for a sink rate of 16.6 feet per second during VTOL landings and 10 feet per second during CTOL landings. A VTOL sink-rate capability of about 14 feet per second was demonstrated during the test program.

STRUCTURE

The aircraft structure consists of the fuselage, wing, and empennage, and the associated control surfaces.

Fuselage

The forward section of the aircraft fuselage contains the forward pitch/yaw reaction control nozzles, an equipment compartment, and the crew compartment. The crew compartment is enclosed by a plastic windshield and a laterally hinged canopy and contains the pilot's instrument panel and controls and an ejection seat. The nose landing gear, when retracted, is housed beneath the pilot's compartment.

The center fuselage section houses the augmented jet ejector system, the fuel tanks, and the engines. Doors are located on both the top and the bottom of the fuselage to permit the entrance of the secondary airflow to the ejector and the exit of the secondary airflow and engine exhaust gases during VTOL flight. The fuel tanks are located between the ejector chambers and below the exhaust manifold. The engine nacelles are located immediately outboard of the ejector compartment and contain the engines, diverter valves, and exhaust nozzles and ducting; they also house the main landing gear when it is retracted. The wings are attached to the fuselage immediately below the engines.

The aft fuselage section contains the aft pitch/yaw reaction control nozzles, an equipment compartment, and a drag parachute.

The entire fuselage is of aluminum-alloy construction with the exception of the ejector compartment lining, which is of titanium-alloy skin backed by steel stiffeners. The main landing gear trunnions are attached to the wing front spar carry-through beam.

Wing

The outer portion of each wing is fully cantilevered and attached to the fuselage structure at the nacelle. Each wing is equipped with a trailing-edge flap, an aileron, and two reaction control nozzles. No fuel is carried in the wing. The outer wing panel is a conventional two-spar box beam of aluminum alloy. The inboard structure, between the attachment points, consists of two titanium-alloy spars through the ejector compartment. The leading edge is nonstructural for primary loads. The trailing-edge flaps are single slotted and are mounted on external hinges below the wing lower surface. The flaps are of single-spar, aluminum-alloy construction and are statically and dynamically balanced. A fabric seal is installed between the aileron leading edge and the wing structure.

The ailerons incorporate a geared tab and are mechanically connected to the reaction roll valves located in each wing-tip pod.

Empennage

The empennage is of the T-type, with the horizontal stabilizer mounted atop the vertical stabilizer. The vertical stabilizer is attached to the aft fuselage frames in a continuous skin connection. The horizontal surface is bolted to the vertical surface. Both surfaces are of two-spar construction with rib-supported skin. The horizontal surface consists of a fixed stabilizer and two interconnected elevators. The elevators are actuated conventionally and are also incorporated with a droop mechanism driven from the right diverter valve. A bullet-type fairing, located at the junction of the horizontal and vertical stabilizers, houses the blowing boundary layer control (BLC) system that is employed on the leading edges of the elevators and horizontal stabilizer during VTOL flight.

AIRCRAFT DIMENSIONAL CHARACTERISTICS

Dimensions and areas of the XV-4A are as follows:

WING

Wing area (total, including theoretical)	104.17 square feet
Span (theoretical)	25.0 feet
Aspect ratio	6.0
Root chord	6.00 feet
Tip chord	2.33 feet
Taper ratio	0.389
Mean aerodynamic chord (MAC)	4.436 feet
MAC leading edge	FS 280.18
Root airfoil section	NACA 64A012
Tip airfoil section	NACA 64A212
Root incidence (with respect to fuselage reference line)	+1.5 degrees
Tip incidence (with respect to fuselage reference line)	-1.5 degrees
Dihedral (at quarter chord)	0 degrees
Sweepback (at quarter chord)	4.18 degrees
Aileron area (per wing)	2.14 square feet
Aileron span (per wing)	2.75 feet
Aileron chord (percent wing chord)	28.0 percent
Aileron travel (VTOL and CTOL)	± 20 degrees

Flap area (per wing)	6.51 square feet
Flap span (per wing)	5.50 feet
Flap chord (percent wing chord)	30.0 percent
Flap travel	0 to 40 degrees

HORIZONTAL STABILIZER

Area (including elevator)	26.45 square feet
Span	10.67 feet
Aspect ratio	4.30
Root chord	3.54 feet
Tip chord	1.42 feet
Taper ratio	0.402
Mean aerodynamic chord (MAC)	2.631 feet
Airfoil section (constant)	0010-2.00 -40/1.575
Incidence (with respect to fuselage reference line)	0 degrees
Dihedral	0 degrees
Sweepback (at quarter chord)	13.0 degrees
Elevator area	5.29 square feet
Elevator span	10.67 feet
Elevator mean chord	0.35 foot
Elevator travel (VTOL)	0 to -60 degrees
Elevator travel (CTOL)	± 30 degrees
Distance from wing MAC quarter chord to horizontal stabilizer MAC quarter chord	14.97 feet

VERTICAL STABILIZER

Area (including rudder)	27.46 square feet
Span (theoretical)	6.08 feet
Aspect ratio	1.35
Root chord	5.96 feet
Tip chord	3.07 feet
Taper ratio	0.515
Mean aerodynamic chord (MAC)	4.668 feet
Airfoil section (constant)	NACA 64A012
Sweepback (at quarter chord)	32.1 degrees
Rudder area	5.19 square feet
Rudder span (along hinge line)	4.71 feet
Rudder mean chord (perpendicular to the rudder hinge line)	1.08 feet

Rudder travel (VTOL and CTOL)	± 20 degrees
Distance from wing MAC quarter chord to vertical stabilizer MAC quarter chord	12.38 feet

FUSELAGE

Length	32.88 feet
Maximum height	5.17 feet
Maximum width	4.67 feet

GENERAL

Height of highest fixed part of aircraft from ground level	11.8 feet
Maximum aircraft length	33.78 feet
Tread of main wheels	6.58 feet
Wheelbase (longitudinal distance from main wheel axis to nose wheel axis)	10.31 feet

WEIGHT STATEMENTS

DESIGN WEIGHT STATEMENT

The design weight statement of the XV-4A is as follows:

Item	Weight - Pounds
Wing group	348
Tail group	229
Fuselage group	1,037
Landing gear group	271
Nacelle group	330
Propulsion group	1,603
Fixed equipment group:	
Surface and reaction controls	423
Instruments	119
Hydraulics	100
Electrical	384

Furnishings	151
EMPTY WEIGHT	4,995
Operating equipment and fuel:	
Crew	200
Oil and unusable fuel	30
Flight test equipment	300
Fuel	1,675
VTOL TAKEOFF WEIGHT	7,200

ACTUAL WEIGHT STATEMENT

The weight statement of the XV-4A as it was actually flown is as follows:

Item	Weight - Pounds
Wing group	350
Tail group	170
Fuselage group	1,207
Landing gear group	291
Nacelle group	245
Propulsion group	1,648
Fixed equipment group:	
Surface and reaction controls	486
Instruments	73
Hydraulics	62
Electrical	376
Electronics	29
Furnishings and equipment	209
Air conditioning	32
EMPTY WEIGHT	5,178
Operating equipment and fuel:	
Pilot and seat pack	232
Oil and unusable fuel	60
Flight test equipment	583
Fuel	1,147
VTOL TAKEOFF WEIGHT	7,200

AIRCRAFT PERFORMANCE CRITERIA

The XV-4A was designed to have a vertical takeoff thrust of 8, 375 pounds under sea level standard conditions. Therefore, at its VTOL design gross weight of 7, 200 pounds, the aircraft would have had a thrust-to-weight ratio of 1. 16. This thrust-to-weight ratio reduces to about 1. 10 after taking into account the downward movement of the air in the vicinity of the aircraft and the lower pressures on the underside of the aircraft (suckdown) that are induced by the VTOL propulsion system.

The maximum vertical thrust actually attained during the XV-4A flight tests was approximately 7, 800 pounds when corrected to sea level standard conditions. This value gives a thrust-to-weight ratio of 1. 08 when the effects of suckdown are neglected.

The XV-4A design moments of inertia for three flight conditions are shown in Table I. These moments of inertia and the aircraft reaction control moments give the VTOL control power shown in Table II. Data in this table, which are based on maximum VTOL propulsion system thrust and a gross weight of 7, 200 pounds, illustrate two interesting factors concerning the XV-4A's VTOL performance. First, the aircraft

TABLE I

DESIGN MOMENTS OF INERTIA

	Center of Gravity		Moment of Inertia (slug-ft^2)			
Condition	FS	WL	I_xCG	I_yCG	I_zCG	Remarks
Minimum Flying Weight, 5, 450 lb	285. 3	100. 4	1, 693	6, 442	7, 260	Maximum zero fuel weight less 175 pounds of instrumentation plus 100 pounds of fuel
Maximum Zero Fuel Weight, 5, 525 lb	285. 1	100. 2	1, 697	6, 885	7, 703	Maximum VTOL take-off weight less 1, 675 pounds of fuel
Maximum VTOL Takeoff Weight, 7, 200 lb	285. 9	96. 2	1, 798	7, 701	8, 439	

TABLE II

VTOL CONTROL POWER
(In Excess of Trim Requirements)

No. of Engines Operating	Control	Flight Mode	BLC	Control Power (ft-lb)	Control Power (radians/sec^2)
Two	Lateral	Hover	On or Off	3, 100	1. 723
One	Lateral	Hover	Off	2, 340	1. 301
One	Lateral	Hover	On	1, 770	0. 984
Two	Longitudinal	Hover	Off	6, 300	0. 818
One	Longitudinal	Hover	Off	3, 150	0. 409
Two	Longitudinal	Transition at 70 knots	On	2, 660	0. 345
Two	Longitudinal	Transition at 70 knots	Off	-2, 740*	-0. 356*
One	Longitudinal	Transition at 70 knots	On	4, 100	0. 532
One	Longitudinal	Transition at 70 knots	Off	-1, 300*	-0. 169*
Two	Directional	Hover	Off	2, 525	0. 299

*Negative sign indicates uncontrollable condition.

does not have adequate pitch control power to transition if the horizontal tail boundary layer control system is not functioning. Second, and this is a phenomenon associated with the large quantity of air flowing through the ejector and the subsequent large trim requirements, the XV-4A has more pitch control power available during transition when only one engine is diverted through the ejector than when both engines are diverted, despite the decrease in the reaction control power.

The design airspeeds for the XV-4A are shown in Table III. The conventional flight tests of the aircraft were conducted below 200 knots equivalent airspeed (KEAS), with the exception of a single flight which was made to 300 KEAS and 2.09g in order to demonstrate the structural integrity of the aircraft at that point in its flight envelope.

TABLE III

DESIGN AIRSPEEDS

Condition	Maximum Airspeed
Design Cruise Speed	350 KEAS or Mach 0.53
Design Dive Speed	450 KEAS or Mach 0.68
Design Flap Speed	200 KEAS up to 25,000 ft
Design Gear-Down Speed	205 KEAS up to 25,000 ft
Design Maneuver Speed	313 KEAS for gr wt of 7,200 lb 272 KEAS for gr wt of 5,450 lb
Ejector-Door-Open Speed	170 KEAS

AIRCRAFT SYSTEMS

FLIGHT CONTROL SYSTEMS

The XV-4A aircraft controls are conventional and consist of a control stick for lateral and longitudinal control and a pair of rudder pedals for directional control. No additional controls are provided for either conventional or VTOL mode flight. Control of the aircraft in conventional flight is by movement of elevator, ailerons, and rudder, which are hydromechanically connected to the control stick and rudder pedals. In the VTOL mode, control of the aircraft is provided by the reaction control systems which utilize turbojet engine compressor bleed and exhaust gases. The reaction control valves are connected directly to the control stick and rudder pedals and cannot be disengaged; thus, no programming or changeover of controls is required as the aircraft progresses from one flight mode to another. In the design of the control systems, use of non-boosted aerodynamic controls in conventional flight and hydraulically boosted reaction controls plus boosted aerodynamic controls in VTOL mode flight was anticipated. In conventional flight, aerodynamic feedback of control surface moments provides all feel forces; in VTOL mode flight, feel forces are provided by double-acting, preloaded spring bungees. The hydraulic boost system was provided to give uniform control force characteristics and was irreversible to prevent stability augmentation system inputs from being transmitted back to the pilot. A summary of control characteristics is given in Table IV.

TABLE IV

CONTROL AXES SUMMARY

Control Axes	Control Displacement	Surface Deflection	Reaction Control Valve Movement	Artificial Feel Force
Longitudinal Elevator Pitch Valve	± 5 in.	± 30 deg	± 1.75 in.	± 8 lb

TABLE IV (Contd)

Lateral Aireron Roll Valve	± 5 in.	± 20 deg	± 0.57 in.	± 7 lb
Directional Rudder Yaw Valve	± 3-1/2 in.	± 20 deg	± 45 deg	± 25 lb
Flaps	Up-Down	0-40 deg	-	-

REACTION CONTROL SYSTEMS

The reaction controls, which are required during hover and transition phases of flight, are distinguished from the conventional aerodynamic control systems in that reaction control valves impart forces to the airplane to provide a control force directly proportional to a control displacement, regardless of airspeed. The major differences between the reaction controls of the three axes are the type of reaction control valve used and the manner in which the valves obtain pneumatic power, as shown in Figure 5.

Pitch/Yaw Systems

Longitudinal and directional control of the aircraft is accomplished by the pitch/yaw reaction control valves shown in Figure 6. The pitch/yaw reaction control valves obtain pneumatic power from the turbojet engine exhaust gases. Exhaust gas is bled from the forward and aft ends of the ejector manifold. Each engine supplies pneumatic power to one pitch/yaw valve located in the nose and one pitch/yaw valve located in the tail of the aircraft. The complete system consists of four rotating valves with two movable vanes each. The valves operate under continuous bleed conditions. The valves are rigged such that in neutral position, the forward and aft valves exhaust equal amounts of gas in a downward direction, thus providing equal thrust on each end of the aircraft.

Pitch Function

For pitch control, movement of the control stick forward closes the front valve vanes and opens the aft valve vanes. Closing of the valve vanes decreases the thrust output, and opening of the vanes increases the thrust output; thus, in the above condition, a nose-down pitching moment is imparted to the airplane. Aft stick motion reverses the above operation such that a nose-up pitching moment is imparted to the airplane without changing the net upward thrust.

Yaw Function

For yaw control, movement of the left rudder pedal forward causes the forward valves to rotate in a direction which results in a reaction force that pushes the nose to the left. Concurrently, the rotation of the aft valves in the opposite direction results in a reaction force that pushes the tail to the right. The combined action is a counterclockwise rotation of the aircraft about its center of gravity. The combined operation of the rudder pedals and the longitudinal stick reduces the commanded pitching moment as a cosine function of the angle of valve rotation.

Roll System

Lateral control of the aircraft is accomplished by use of the roll control valves located in each wing-tip pod (see Figure 7). Each pod contains two roll valves, one directed upward and the other directed downward. The roll control system is supplied pneumatic power from the ninth-stage compressor bleed ports of both turbojet engines. A common manifold interconnects the bleed ports of the two engines and provides the ducting for the control valves at each wing tip. Roll reaction control is accomplished by movement of a plug-shaped sleeve into the valve exhaust area opening; this movement modulates the valve thrust output. The roll reaction valves operate only upon demand to minimize consumption of compressor bleed gas. With the control stick in neutral, no gas is being exhausted from the valves. Movement of the stick to the left opens the upward-pointing valve at the left wing tip and the downward-pointing valve at the right wing tip. The other valves remain closed through a system of one-way spring cartridges, and the resulting moment rolls the aircraft to the left. Movement of the stick to the right reverses the functions. Initial design of the pneumatic system isolated bleed gas from each engine; however, a requirement for additional roll control power dictated that upward-exhausting roll valves be installed. The addition of two more roll valves created a requirement for more compressor bleed gas flow, and an additional line was added that bypassed the interconnecting manifold. This, then, resulted in the flow of all the bleed gas from one engine to the other during engine start, and a subsequent overtemperature condition, unless the roll valves were opened by movement of the control stick.

System Gearing Relationships

For the reaction control system on all three axes, the gearing relationship between the pilot's control and the reaction control valves is such that a control movement corresponding to one-half the maximum displacement will give 70 percent of the reaction control moment. This relationship, shown in Figure 8, provides 70 percent of the control power to overcome hard-over stability-augmentation-system failures.

AERODYNAMIC CONTROL SYSTEMS

The aerodynamic control systems are comprised of conventional cable and bell-crank/push-rod systems, mechanical and electrohydraulic servo systems, and electrical trim systems. The aerodynamic control surfaces consist of conventional elevator, ailerons, and rudder. These controls deflect normally throughout VTOL mode flight and are required at higher VTOL mode flight speeds to supplement the reaction controls.

Longitudinal Control System

The aerodynamic longitudinal control system incorporates three features which are required for longitudinal control during VTOL mode flight: an elevator drooper, a horizontal stabilizer and elevator boundary layer control system, and an elevator down spring.

Elevator Drooper

The elevator drooper is a mechanical actuator which repositions the elevator with respect to the stick. When the drooper is activated, the elevator position, with the control stick at neutral, droops from the faired position (0 degrees) to 30 degrees down. The deflection of the elevator is ± 30 degrees from either the drooped or the nondrooped neutral position. Actuation of the drooper is automatic with diversion of the right engine to the lifting mode. The repositioning of the elevator is required to provide a nose-down pitching moment, in conjunction with the reaction controls, to balance the nose-up pitching moment (momentum drag moments) which increases as forward velocity increases.

Boundary Layer Control System

The boundary layer control system is required during transition flight to aid in overcoming the inherent pitch-up moment of the ejector concept. The flow of relatively high energy air over the horizontal stabilizer and elevator delays airflow separation at large elevator deflections and thereby increases the tailplane lift and, consequently, increases the nose-down pitching moment. The boundary layer control system, which is activated through the flight mode selector switch (FMSS), obtains bleed gas from the manifold which interconnects bleed ports of the turbojet engines. The maximum bleed gas requirement of the boundary layer control system is 1.5 pounds of gas per second at 70 psia and 500° F. Movement of the FMSS to the VTOL position opens the inlet valve; this permits ram air to pass into the ejector bullet and out through slots in the horizontal stabilizer ahead of the elevator. When the inlet valve is fully open, a BLC shutoff valve in the bleed line is opened. Mixing of the high-temperature bleed gas with ram air reduces the temperature of the air passing over the stabilizer and elevator. A spring-loaded button switch, located on the right engine throttle lever, permits closing of the BLC shutoff valve to allow an increase in roll control power or an increase in thrust.

Elevator Down Spring

The elevator down spring was not originally intended for the elevator system, but was later installed to offset the high elevator-up hinge moments encountered in VTOL mode flight. The hydraulically actuated down spring is connected to the left engine diverter valve switch. In conventional flight, the down spring actuator solenoid is deenergized; this interconnects the two ends of the hydraulic cylinder and allows the actuator to bypass fluid as it is moved. During Phase I VTOL flight, the rod end is pressurized to provide the down elevator moment. No force or movement is observed at the control stick as the elevator is moved by the pilot. Elimination of the down spring was planned by installation of a larger area hydraulic boost actuator designed to cope with the elevator hinge moments.

Lateral Control System

Aerodynamic lateral control, which is effected by ailerons, is conventional in all respects. However, the lateral control system has been incorporated with an aerodynamic seal, a geared or servo tab, and relatively frictionless bearings to reduce the system aerodynamic feedback and friction forces.

Directional Control System

Aerodynamic directional control is effected by a conventional rudder. The rudder system is conventional in all respects, and no VTOL mode requirements are placed upon this control system.

TRIM SYSTEMS

Two trim systems are installed in the aircraft: one is utilized for conventional flight and operates aerodynamic control surfaces; the second is used during VTOL flight in conjunction with the artificial feel system. Selection of the proper system is automatic with operation of the flight mode selector switch. Operation of the two trim systems is identical. The switch for longitudinal and lateral trim is mounted on the control stick grip, and the switch for directional control is located on the left console. The authority of all trimming devices is limited to one-half the corresponding pilot command to allow sufficient control travel to override a runaway trim malfunction.

Conventional Trim

For conventional flight trim, operation of the trim switch positions an electromechanical actuator mounted within the control surfaces, which in turn moves a trim tab to the desired position. Trim tab position indicators for all three axes are located on the instrument panel.

VTOL Trim

The VTOL flight trim system is energized when the flight mode selector switch is in the VTOL position. This also deenergizes the conventional trim system. VTOL trim commands are transmitted from the appropriate control switch to the feel trim actuator. Corresponding movement of the actuator is in the direction that will relieve the feel spring forces.

Alternate VTOL Elevator Trim

An alternate VTOL pitch trim switch, located on the side console, is provided to permit selection of the elevator trim tab position with the aircraft in the VTOL mode. This switch is provided to reduce the stick force discontinuity which may occur when shifting from hydraulically boosted controls operation to manual controls operation.

FLAPS

The wing flaps are single-slotted, trailing-edge flaps and are powered by hydraulic cylinders located in each engine nacelle. The cylinders operate the surfaces through a push-pull rod system, which also mechanically interconnects the two surfaces. Flap position (two positions) is selected by a lever located on the instrument panel. Failure of the electrical or hydraulic systems hydraulically locks the flaps in the selected position.

PROPULSION SYSTEM

Two Pratt and Whitney JT-12 turbojet engines provide power for either conventional or VTOL flight. The engines are installed in nacelles along the side of the fuselage and above the wing. Each engine is equipped with a diverter valve assembly to direct the exhaust gas out of the conventional tailpipe for CTOL flight or into the ejector manifold for VTOL flight.

JT-12A-3 (LH) TURBOJET ENGINE

The JT-12A-3(LH) turbojet engine has a nine-stage, single-rotor compressor driven by a two-stage reaction turbine. A can-annular combustion chamber contains eight burner cans into which fuel is sprayed through single, dual-orifice nozzles mounted at the inlet of each can. High-energy ignition units and igniter plugs are used to start combustion. A hydromechanical fuel control governs the compressor rotor speed, schedules fuel flow to provide the thrust called for by the throttle setting in the cockpit, and automatically compensates for conditions at the compressor air inlet. An air-bleed valve opens automatically to bleed interstage compressor air overboard to facilitate engine starting and operation at low thrust. This engine has a maximum uninstalled thrust rating of 3,300 pounds under sea level standard day static conditions.

Air Induction System

The engine air induction system consists of a conventional, fixed, round inlet in each nacelle forebody and is designed to give good inlet recovery from static operation to the maximum limit speed. The exhaust system of each engine consists of a diffuser duct attached to the engine flange, with a bellows attachment that permits differential expansion and a diverter valve that directs the exhaust gas flow.

Diverter Valve

Each engine is equipped with a hydraulically actuated diverter valve that consists of two interconnected, moving doors, one curved and the other straight. In conventional flight, the curved door blocks the duct to the ejector manifold and the straight door is faired into the exhaust stream. In the VTOL mode, the straight door blocks the conventional tailpipe and the curved door provides a turning vane.

LIFT SYSTEM

For vertical thrust, the XV-4A uses augmented deflected exhaust gas from the two turbojet engines.

System Description

The lift system comprises two tapered manifolds, each connected to the inboard face of the respective diverter valve. Each manifold is equipped with a series of 10 transverse nozzle bars, each of which has a slotted nozzle array directed downward into a mixing chamber. The two mixing chambers are aligned longitudinally with the fuselage and are offset laterally from the aircraft plane of symmetry, one on each side. Vertically, the chamber center lines diverge from top to bottom with respect to one another. Each chamber has parallel walls over the initial 50 percent of its depth and divergent walls over the remainder. The greater portion of the outer divergent wall is formed by the closure door, which has an adjustable setting to permit attainment of optimum secondary exit nozzle area. Each chamber is also equipped with a pair of inlet closure doors, with the outer door forming the curved inlet contour. The design ratio of the ejector outlet area to the primary nozzle area for the XV-4A was 13.6. Figure 9 is a cutaway top view of the fuselage section and depicts the arrangement of the main components of the propulsion system. The two JT-12 engines are mounted just outboard of the fuselage above each wing root. Aft of each engine, there is a diverter valve which is the single major moving part on the lift system. Inboard of the engines is the main manifold leading to nozzle branches, with the nozzles directed downward into the mixing chambers. The nozzle branches are inclined rearward at an angle of 12 degrees to the aircraft vertical reference plane.

Diverter Gas Flow

Figure 9 also illustrates how exhaust gases from the two turbojet engines may be directed aft for normal flight or into the manifold for VTOL flight. The upper engine diverter valve in this figure is positioned for conventional flight and the lower valve is positioned for VTOL flight. In the diverted or VTOL position, the exhaust gases are directed through the diverter valve into the manifold and from there through the nozzle branches into the mixing chamber. The exhaust gases flow directly through the mixing chamber and out the exit at the bottom, as shown in Figure 10. Thrust augmentation is obtained by inducement of secondary airflow into the ejector system. The high-velocity, low-mass-flow primary gases mix with the low-velocity, high-mass-flow secondary air to result in an increase of momentum in the mixed gases. The inducement (or jet pump effect) of secondary air into the system increases the mass flow by about five and one-half times. Each engine supplies exhaust gas at approximately 1200° F to alternate nozzle branches so that there is no tendency for the aircraft to roll or pitch when the two engines are at different power levels or if the thrust from only one engine is being directed through the mixing chamber.

System Losses

As can be expected, the vectoring of gases through the ducting results in pressure losses, which in turn results in the loss of basic, unaugmented thrust. Calculations based on actual system hardware indicate a total pressure loss of 0.84 percent in the diffuser, of 1.48 percent in the diverter valve, of 3.87 percent in the nozzle manifold and inlet, and of 1.37 percent in the nozzle ducts. Total calculated pressure loss is 9.02 percent. This pressure loss represents a thrust loss, before augmentation, of approximately 5 percent. As previously indicated, the introduction of cooler outside air reduces the temperature and velocity of the resultant airflow. The temperature of the exhaust gas is reduced from 1200° F at the engine exhaust to approximately 300° F at the lower door exits.

STABILITY AUGMENTATION SYSTEM

A stability augmentation system has been incorporated into the aircraft to provide artificial stability and to improve handling characteristics during hover and transition flight. The system utilizes electromechanical transducers to sense aircraft responses, electronic computers and amplifiers to convert and amplify the sensor outputs to the desired level, and electrohydraulic servos to power the airplane controls. The servos operate through the same mechanical system as pilot-initiated commands to actuate the reaction and aerodynamic controls. The SAS inputs are in series with the pilot commands, and the stick will not move as a result of these inputs. The aerodynamic and reaction controls are mechanically linked, and they operate in unison. This permits the aerodynamic controls to add effectiveness to the reaction controls. The authority of the SAS is limited to 50 percent of the stick authority, based on safety considerations in the event of a servo malfunction. This limiting allows the pilot to retain approximately 70 percent of the available reaction control moment to override the servo and to maintain aircraft control while the defective system is being disengaged. Essentially, two SAS configurations were utilized during the XV-4A development program. The configurations were selected as a result of analytical investigation and as a result of flight test.

AIRCRAFT STABILITY SYNTHESIS

An autostabilization system synthesis and analysis study was conducted by the Eclipse-Pioneer Division of the Bendix Corporation to determine the XV-4A stability characteristics and augmentation requirements. The results of this study indicated that longitudinally the XV-4A is inherently unstable under all Phase I and Phase II flight conditions and that laterally the XV-4A is inherently unstable under almost all Phase I and Phase II flight conditions. Directionally, the XV-4A is essentially stable.

Longitudinal Axis

The free airframe longitudinal instability was determined as severe and divergent with no distinct short-period or phygoid modes. The time to double amplitude approaches 1 second at some flight conditions. However, it is conceivable that the aircraft could be flown without longitudinal stability augmentation.

Lateral Axis

The lateral instability exhibited by the XV-4A is primarily an interchange between roll angle and sideslip, which produces a divergent oscillation about the axis. Very little coupling between the roll and yaw axes was noted. The lateral instability is most severe at low Phase I speeds, and it is doubtful that the aircraft could be flown without lateral stability augmentation.

Directional Axis

The directional axis exhibited no unusual characteristics like the other axes, and the aircraft responded as a pure inertia load.

INITIAL SAS CONFIGURATION

As a result of the system synthesis and analysis, the effects of various forms of feedback on the instabilities were examined to determine the improvement that could be made in handling.

Longitudinal Control System

Initial design for the longitudinal control system included pitch rate feedback for stability, pitch attitude feedback to aid in the ability to hold a desired attitude, and an attitude command signal, derived from a stick force sensor, to increase the effective stick sensitivity. Also included in this system was a command modifier to improve the system response to large, rapid stick inputs and a pitch attitude synchronizer to enable the command of large trim changes in pitch attitude without saturating the pitch damper servo.

Lateral Control System

Design for the lateral control system included roll rate feedback for stability augmentation and roll attitude feedback for roll stabilization during transition flight. A roll attitude command signal, via the stick force sensor, and a command modifier were included for use in conjunction with the roll attitude feedback. Unlike the pitch channel, a synchronizer is unnecessary because stick deflections required to trim roll are negligible.

Directional Control System

The designed directional control system consisted of rate damping with no attitude functions. However, lateral acceleration feedback was incorporated to aid the aircraft in response to lateral gust disturbances in the yaw channel.

FINAL SAS CONFIGURATION

With the aircraft in the initial SAS configuration during a flight, servo saturation was encountered on the lateral axis. This resulted in the loss of all stability functions on the lateral axis and in a subsequent hard landing. Inasmuch as some flight testing had demonstrated that handling qualities were acceptable using a rate-only feedback, all attitude commands and the lateral accelerometers were electrically disconnected from the system, and a rate-only configuration was assumed.

SAS VARIABLE GAIN SYSTEM

A manually adjustable gain setting feature was built into the SAS to permit evaluation of VTOL handling qualities. Even though analysis had shown that fixed gains were suitable for all phases of transition, adjustable gain to permit rapid changes after a series of flights was included. This permitted a determination of gain settings that provided the optimum combination of stability and maneuverability.

SAS REDUNDANCY

Because of the basic aircraft instabilities, some redundancy had to be incorporated into the system. The redundancy selected was dualization of the components, with pilot monitoring of the system and manual switching required. It was felt that the period of divergence, in the event of a SAS failure, would be great enough to permit manual selection of the alternate system. Both systems were complete and identical in function and operation, and no switching transients were encountered.

MISCELLANEOUS AIRCRAFT SYSTEMS

ELECTRICAL SYSTEM

The electrical system consists of a primary, two-bus system and a secondary, three-bus system. The primary system consists of a 28-volt d-c system utilizing two 30-volt d-c starter-generators and one 24-volt battery as supply sources. The secondary system consists of a 115-volt a-c system utilizing inverters for power conversion requirements of the a-c bus loads.

HYDRAULIC SYSTEM

The 1,500-psi hydraulic system is powered by two engine-driven variable-volume pumps. Each pump is capable of supplying power to operate any two hydraulic subsystems. The systems operated hydraulically are the flaps, landing gear, ejector doors, diverter valves, and stability augmentation system actuators.

ESCAPE SYSTEM

The aircraft is equipped with an ejection seat designated the Douglas Aircraft Company Escapac I-C. The seat is designed to permit successful ejection at any altitude and speed in the aircraft's flight envelope, including ground-level altitude and zero speed (zero-zero). It is equipped with a fixed track and a rocket catapult. The upper part of the seat is designed to break through the plastic canopy.

FLIGHT TEST PROGRAM

The XV-4A flight tests were conducted at Dobbins Air Force Base, Marietta, Georgia. The XV-4A aircraft made 82 hover flights, for a hover flight time of about 8-1/2 hours. A total of 69 conventional flights were made, for a flight time of about 28-1/2 hours. The total flying time during the test program was about 37 hours.

For VTOL flights, the ambient atmospheric conditions were taken into consideration, and the aircraft fuel load was reduced until the aircraft thrust-to-weight ratio was about 1.00. Some short lift-offs were made at a thrust-to-weight ratio of about 1.03 (neglecting suckdown). The VTOL performance improved, of course, as the fuel supply was consumed. For the seven transitions made from the runway, the flight test instrumentation was removed from the aircraft and the fuel load was reduced.

The primary purpose of the XV-4A program was to determine the feasibility of the augmented jet ejector concept for VTOL aircraft. Consequently, the accumulation of accurate quantitative flight test data was not emphasized. While the data presented here represent the true XV-4A characteristics, the absolute value of some traces is questionable because of possible errors in ground zeros. Also, the data are not corrected for any phase errors that may exist between traces.

HOVER FLIGHT

Because of a failure to achieve the predicted vertical lift from the augmented jet ejector, the hover capability of the XV-4A was somewhat below that expected. The ejector manifold used during the early hover flight was designated the B manifold. Its VTOL performance is shown in Figure 11. The manifold installed in the aircraft during the latter portion of the flight tests, including the transitions, was designated the D manifold, and its VTOL performance is shown in Figure 12.

TAKEOFFS

Hover takeoffs were accomplished by opening the ejector doors, placing the diverter valves in the VTOL position, and rotating the aircraft to a nose-high attitude while increasing the engine thrust until the aircraft left the ground. The aircraft usually hovered at a nose-high attitude of 10 to 12 degrees, as shown in Figures 13A and 14A. Since takeoff usually required all the available power, height control, which was a function of throttle position, was initially nonexistent but improved as fuel was consumed and the thrust-to-weight ratio increased. In general, the vertical lift was marginal.

LANDINGS

Hover landings were accomplished by reducing power and letting the aircraft descend, then increasing power to arrest the descent and touching down on the main gear. Some landings, mainly those with a slight forward translational velocity, included flares in order to provide the maximum vertical thrust component and to stop the translation.

STABILITY AND CONTROL

Pitch control step inputs were made, but only small deflections were used. Pitch accelerations and velocities were satisfactory. During these step inputs, there was a noticeable loss of altitude as the pitch attitude changed. However, when pitch changes were slow and small, it was possible to move forward without losing altitude. Without the SAS, the aircraft was essentially stable longitudinally.

Roll control inputs were very small, and only very small roll velocities were attained. The control power was satisfactory, and no loss in altitude was noted during the inputs. During hover, the aircraft was laterally unstable, as indicated by the continuous SAS inputs as shown in Figures 13B and 14B. Sideward flight was very steady and easy. A time history of a lateral stick release from hover is shown in Figure 15. During sideward translational flight, the aircraft had inherent positive dihedral effect as a result of the ejector airflow momentum drag.

Yaw control power was considered to be satisfactory, with very little height lost during the yawing motion. The initial yawing acceleration was considered to be a little slower than that of most single-rotor helicopters, but after about 2 seconds, the velocity increased to an estimated 10 to 15 degrees per second. The aircraft was essentially stable directionally, as indicated by the limited SAS inputs as shown in Figures 13C and 14C.

The SAS is required for hover flight owing to the lateral instability. The SAS contribution during hover flight was excellent, and hover control was easy to maintain.

The stick and rudder control forces during hover were reported by the pilots to be rather light because of the control boost which was employed. The VTOL control system is further discussed in another portion of this report.

GROUND EFFECT

The ground effect was noticeable from the ground to a wheel height of about 3 feet as a high-frequency vibration (rumbling) in the airframe. The intensity of the vibration decreased as altitude was gained and was hardly noticeable above a wheel height of 3 feet. A definite change was noted in the ground effect during flights on windy days. When the aircraft was headed into the wind, the airframe vibration was relatively low. As the crosswind component was increased, the airframe vibration increased.

The power of the engines was observed to increase as the aircraft moved out of ground effect. This can be attributed to lower engine inlet temperatures out of ground effect because of a decrease in reingestion.

PHASE I FLIGHT

In the Phase I flight regime, both engines were in the lifting mode, and the aircraft had a forward translational velocity of 0 to approximately 80 knots.

The aircraft was translated forward from hover by slightly lowering the nose to provide a horizontal thrust component. Because of the limited vertical thrust, the aircraft usually settled back to the runway one or more time (leapfrogged) as it gained forward speed. This leapfrogging is shown in Figure 16A. After attaining a speed of approximately 20 or 30 knots, the aircraft was less affected by reingestion and, according to the pilot, "accelerated or climbed fairly well with the feeling of some excess power."

As forward speed increased, the nose of the aircraft was continually lowered until it was approximately 10 degrees below the horizon. With the fuselage in this attitude, the aircraft accelerated to its maximum Phase I speed of approximately 80 knots. The data in Figures 16A, 16B, and 16C were recorded during a translational flight along the runway.

Deceleration in Phase I was obtained by using a flare technique while maintaining altitude by throttle manipulation.

During translational flight along the runway, the pilot monitored the aircraft longitudinal attitude. During the VTOL mode flights at high altitude, the pilot monitored the angle of attack, since the aircraft could not maintain constant altitude.

TRANSITION FROM PHASE I TO PHASE II FLIGHT

Transition from Phase I to Phase II flight was made at approximately 80 knots and involved changing the number 1 engine from the lifting mode to the thrusting mode. The number 2 engine remained in the lifting mode throughout Phase I and Phase II flight.

Data for an in-flight transition from Phase I to Phase II are presented in Figures 17A, 17B, and 17C. In order to reach Phase I flight mode, the aircraft backed down from conventional flight through Phases III and II and into Phase I. During the time that the data were recorded, the aircraft had a rate of descent of about 1,000 feet per minute. The engines were producing maximum power during the transition.

Figure 17A indicates that the pilot started raising the aircraft's nose, to minimize the anticipated increase in rate of descent, shortly before initiating the transition. At the transition, the longitudinal acceleration was increased as the vertical acceleration was decreased. The elevator down trim requirement was about 10 degrees less in Phase II than in Phase I.

Figures 17B and 17C show a unique characteristic of this aircraft. Following transition from Phase I to Phase II, the lateral thrust component of the number 1 engine was countered by the aircraft's assuming a left-wing-down attitude of about 5 degrees. In this condition, the aircraft maintained flight straight ahead without either sideslip or yaw. The apparent increase in lateral acceleration, shown in Figure 17B, was due to the change in orientation of the lateral accelerometer rather than to lateral forces on the aircraft.

TRANSITION FROM PHASE II TO PHASE I FLIGHT

Transition from Phase II to Phase I flight was made at approximately 100 knots and involved changing the number 1 engine from the thrusting mode to the lifting mode. The number 2 engine remained in the lifting mode.

Data for an in-flight transition from Phase II to Phase I are presented in Figures 18A, 18B, and 18C. Figure 18A shows a longitudinal deceleration after the transition and also indicates that the pilot commanded a nose-down attitude. Figures 18B and 18C show the characteristic left-wing-down condition in Phase II changing to essentially a wings-level condition following the transition into Phase I. The slight right-wing-down attitude in Phase I can be attributed to pilot technique.

STABILITY AND CONTROL IN PHASE I FLIGHT

Stability and control about all axes were satisfactory in Phase I, although the directional stability was quite low. The aircraft was very sensitive to gusts in all flight regimes and configurations. During Phase I flight, lateral-directional coupling was noted. However, this motion was easily controlled with the rate-only SAS.

The aircraft positive angle of attack was limited to less than 10 degrees in VTOL translational flight in order to avoid possible pitch-up problems. A limit to the negative angle of attack was not specified. The angle of sideslip was limited to 5 degrees left or right in VTOL translational flight.

The control power requirements in VTOL flight were most critical at about 70 knots in Phase I flight because of the large trim requirements of the jet ejector. Transition into Phase II flight reduced the trim requirements and thereby increased the control power available for maneuvering, even though the aerodynamic control power had only slightly increased and the reaction control power had decreased by one-half. In order to maintain longitudinal control in Phase I flight at 70 knots, the horizontal tail BLC system had to be operating.

PHASE II FLIGHT

In the Phase II flight regime, the number 1 engine was in the thrusting mode, and the aircraft had a translational velocity of from approximately 80 knots to approximately 120 knots.

Acceleration in Phase II was best obtained by reducing power on the number 2 engine in order to reduce the ejector momentum drag. The engine pressure ratio (EPR) of the number 2 engine was usually about 1.5 by the time transition into Phase III was initiated. Deceleration was obtained by decreasing the power on the number 1 engine and increasing the power on the number 2 engine.

In transitioning to Phase II flight, the reason for diverting the number 1 engine to the thrusting mode rather than the number 2 engine can be understood by referring to the aircraft inboard profile, Figure 3, and the ejector performance data presented later in this report. The number 1 engine supplied exhaust gases to the most forward manifold nozzle arms. Hence, diverting this engine to the thrusting mode, while retaining the number 2 engine in the lifting mode, moved the ejector thrust vector slightly rearward and increased the aircraft nose-down moment. This reduced the nose-down longitudinal trim requirement during this phase of flight.

TRANSITION FROM PHASE II TO PHASE III FLIGHT

Transition from Phase II to Phase III flight was made at approximately 120 knots and involved changing the number 2 engine from the lifting mode to the thrusting mode. Data for an in-flight transition from Phase II to Phase III (and continuing on through Phase III to conventional flight) are presented in Figures 19A, 19B, and 19C. This was one of the initial transitions, and the pilot technique shown here was refined somewhat later.

Figure 19A indicated that the aircraft was in a state of descent in Phase II as the transition was initiated. As the transition into Phase III was completed, the rate of descent initially increased but was corrected by the application of up elevator. The elevator trace also shows that the elevator reverted to its conventional position as the elevator drooper was deactivated by the number 2 engine diverter valve's moving to the thrusting mode position. The light control forces indicated that the controls were boosted during the transition.

Figure 19B shows the characteristic left-wing-down attitude in Phase II and the return to essentially wings-level attitude in Phase III. The right-wing-down attitude assumed after about the 10th second of the time history was pilot induced, as shown by the aileron trace. The ejector doors were closed by the start of the 13th second of the time history, as indicated by the slight increase in lateral-directional stability (Figure 19C).

TRANSITION FROM PHASE III TO PHASE II FLIGHT

Transition from Phase III to Phase II flight was made at about 120 knots and involved changing the number 2 engine from the thrusting mode to the lifting mode. Data for an in-flight transition from Phase III to Phase II are presented in Figures 20A, 20B, and 20C.

Figure 20A indicates that the aircraft pitched down rather sharply as the transition was initiated. This was apparently caused by the elevator's being moved down to its drooped VTOL position as the diverter valve for the number 2 engine moved to its lifting mode position. The pilot corrected the pitch down by moving the control stick rearward.

The aircraft did not assume its usual left-wing-down attitude in Phase II because of the efforts of the pilot to keep the wings level by application of the left rudder. This resulted in the aircraft's sideslipping to the right. This sequence of events is shown in Figures 20B and 20C.

Transitions from Phase III to Phase II flight were usually initiated with both engines at an EPR of about 1.5. The pilot usually noted a slight pitch down, left roll, ballooning of the aircraft and a longitudinal deceleration as the number 2 engine was diverted.

STABILITY AND CONTROL IN PHASE II FLIGHT

In Phase II flight, the aircraft maintained a left-wing-down and zero-sideslip flight attitude that required little pilot attention. Climbs were easy to achieve in Phase II flight, and handling appeared normal during shallow turns in each direction. No large trim changes or transients were noted during this phase of flight. The aircraft accelerated as the number 2 engine EPR was decreased and decelerated as it was increased.

PHASE III FLIGHT

In the Phase III flight regime, both engines were in the thrusting mode, both top and bottom ejector doors were open, and the aircraft had a forward velocity of from approximately 120 knots to a maximum allowable velocity of 170 knots.

TRANSITION FROM PHASE III TO CONVENTIONAL FLIGHT

A transition from Phase III to conventional flight consisted of closing the ejector doors. Data for a transition are included in Figures 19A, 19B, and 19C. These data indicate that the only effect on the aircraft in transitioning from Phase III to conventional flight was a slight increase in the lateral-directional stability.

TRANSITION FROM CONVENTIONAL TO PHASE III FLIGHT

A transition from conventional to Phase III flight consisted of opening the ejector doors. Data for a transition are presented in Figures 21A, 21B, and 21C. Little or no trim changes were evident during these transitions, and the only effect on the aircraft was a slight decrease in lateral-directional stability.

STABILITY AND CONTROL IN PHASE III FLIGHT

In Phase III flight, the investigated aircraft characteristics were essentially the same as those in conventional flight, except for the slightly lower level of lateral-directional stability. With the SAS off in Phase III flight, random directional wandering of the aircraft was noted. This wandering of the aircraft was reduced, but still apparent, with the SAS on and was aperiodic with sideslip angles of 5 degrees both left and right. The aircraft also exhibited some oscillatory yawing motions during turns. The aircraft acceleration characteristics were good.

TRANSITION TECHNIQUE

Seven complete transitions from hover to conventional flight to hover were made at low altitude during the XV-4A flight test program. The technique for transitioning from VTOL flight to conventional flight is as follows:

1. Make a vertical takeoff, accelerate to about 30 knots in Phase I configuration, retract landing gear, and continue to accelerate to about 75 knots.

2. Pitch up to 5 degrees nose-up attitude and place the number 1 engine in thrusting mode.

3. Decrease the number 2 engine EPR to 1.5 while accelerating to about 120 knots in Phase II configuration.

4. Place the number 2 engine in thrusting mode, close ejector doors, and retract flaps while continuing to accelerate and establish a climb.

The technique for transitioning from conventional flight to VTOL flight is as follows:

1. Below 170 knots, open ejector doors and lower landing gear and flaps.

2. At about 145 knots, with both engines at an EPR of 1.5, place the number 2 engine in lifting mode.

3. Decelerate to about 90 knots, pitch nose down 5 degrees, and place the number 1 engine in lifting mode while manipulating throttles to control altitude and to ensure that sufficient engine bleed gases are available to provide the necessary reaction control power.

4. Decelerate to a hover and make a vertical landing.

CONVENTIONAL FLIGHT

PROPULSION SYSTEM

The static performance of the conventional propulsion system is presented in Figure 22. In-flight engine shutdowns and restarts were accomplished with ease. The air start envelope, presented in Figure 23, shows the Pratt and Whitney estimated envelope and several points that were demonstrated during the flight tests. All air starts were "quick and cool." Single-engine cruising flight was easily performed.

PERFORMANCE

The XV-4A had excellent acceleration and deceleration with changes in power and impressive climb performance and single-engine characteristics.

STALLS

In both the cruise and the landing configuration, stalls occurred rapidly, without warning, with rapid left roll and downward pitch. However, recovery was easily made by the application of power and aileron. Data for a stall in the clean configuration are presented in Figures 24A, 24B, and 24C; data for the landing configuration are presented in Figures 25A, 25B, and 25C. Figure 26 presents the stalling speed as a function of aircraft gross weight.

TAKEOFFS AND LANDINGS

The XV-4A accelerated rapidly after the application of power for takeoff. At a gross weight of 7,200 pounds, the aircraft required a ground roll of about 1,600 feet for takeoff. Power had to be reduced shortly after lift-off to prevent the maximum speed to which the aircraft had been restricted for flight test purposes from being exceeded.

Power was used throughout the landing approach to control the rate of descent. After touchdown, the ejector doors were opened and both engines were placed in the lifting mode in order to decrease the landing roll. Using this method, the aircraft usually had a landing ground roll of about 3,500 to 4,000 feet. A drag parachute was installed on the aircraft but was not used during the flight tests.

FLAP AND LANDING GEAR TRANSIENTS

A nose-down pitching moment was incurred with the lowering of either or both the flaps and the landing gear, and the aircraft tended to wallow because of the slight decrease in the lateral-directional stability. The opposite effect was noticed during the retracting of the flaps and landing gear. In both cases, the transients and trim changes were easily controlled by the pilot.

STABILITY AND CONTROL IN CONVENTIONAL FLIGHT

Longitudinal

The longitudinal static stability of the XV-4A aircraft with boosted controls is shown in Figure 27. The data indicate slightly positive to neutral stick-fixed static longitudinal stability in the cruise configuration. The longitudinal control effectiveness is shown in Figure 28. Abrupt pullups to 1.8g were made. Upon control release, the resulting oscillation was dead beat, with the aircraft returning to very near the original trim attitude, in a slight left bank, in one-half cycle. The stick forces were very light with the boost on, with all feel being provided by the spring trim package.

Lateral-Directional

The lateral-directional stability of the aircraft is shown in Figures 29, 30, and 31. These data indicate that the lateral-directional stability was positive with a low gradient. Rudder kicks and sideslip releases resulted in the aircraft's making two lateral-directional cycles of approximately equal magnitude and then abruptly stopping in a zero sideslip condition with the left wing down about 25 to 30 degrees. This indicated a strong spiral instability to the left. The aircraft had weak stick-free lateral-directional stability. The lateral control sensitivity was excessive, and the aircraft was extremely sensitive to gusts. In general, the SAS was desirable in conventional flight.

VTOL LIFT IMPROVEMENT PROGRAM

As a consequence of the marginal VTOL performance of the XV-4A aircraft, a program was conducted to improve the lifting capability of the augmented jet ejector. This program, which was not completed until after the flight tests had been concluded, consisted of static ground tests of 2 manifolds (D and F) and a total of 16 ejector configurations.

Six ejector configurations were tested using the D manifold. These ejectors differed in the ejector exit arrangement. Some of these ejectors utilized curved exit outer doors, some of which included fairings between the exits, and some added curved exit inner doors between the two ejector exits. Two of the configurations (designated I and II) are discussed on the following pages.

Ten ejector configurations were tested using the F manifold. These ejectors included variations to the ejector exit arrangements similar to those in the D manifold tests. In addition, the F manifold tests included variations of the ejector inlet geometry and a redesign of the ejector bay splitters. Three of the configurations (designated III, IV, and V) are discussed on the following pages. A description of the entire test program is presented in Reference 10.

The manifolds are shown in Figures 32 through 35, and the test rig used in the program, with various manifold and ejector combinations installed, is shown in Figures 36 through 41. The ejectors were tested in an inverted position in order to provide data that were not influenced by ground effect. The engines (No. 1 and No. 2) used in the data presented here refer to the respective engines (No. 1 and No. 2) on the aircraft. The data presented here have been corrected to sea level standard conditions where applicable.

A comparison of the ejector system augmentation ratios for all five of the tested configurations is presented in Figure 66.

D MANIFOLD

The D manifold, shown in Figure 32, and ejector Configuration I, defined in Figures 42, 43, 44, and 45, constitute the manifold and ejector combination used during the latter portion of the aircraft flight tests. This combination was tested initially on the test rig and was used as a standard for the remaining configurations tested during this program.

The first test was conducted to determine the thrust available at the nozzle exits of the D manifold. To do this, the bare D manifold was installed on the test rig, as shown in Figures 36 and 37. Then the manifold nozzle thrust, resultant force inclination, and resultant force displacement were determined as functions of engine pressure ratio for each engine alone and for both engines combined. These data, which were obtained from test rig load cell data, are presented in Figures 50, 51, and 52. The manifold nozzle thrust is defined as the thrust at the manifold nozzle exits. The resultant force inclination is defined (with respect to the XV-4A aircraft) as the angle of inclination of the ejector force vector with respect to a line perpendicular to the aircraft fuselage reference line, with the angle being positive if the vector is inclined toward the nose of the aircraft. The resultant force displacement is defined (again with respect to the XV-4A aircraft) as the displacement of the ejector force vector with respect to the 0.10 MAC point, with the displacement being positive if the vector is forward of this point. Extrapolating the test data to sea level standard conditions and adjusting it for the differences between the test rig engines and the aircraft engines gave a D manifold thrust of 5,650 pounds at an EPR of 2.28. This 5,650-pound value included the effects of engine inlet pressure recovery, power extraction, compressor and exhaust gas bleeds, et cetera, but did not include the lifting contribution of the reaction control system.

CONFIGURATION I

Ejector Configuration I was installed on the test rig along with the D manifold. This combination provided the best flight performance of any ejector and manifold combination used in the XV-4A aircraft during the VTOL flight tests. Data for this combination were obtained as a function of EPR and ejector exit door angles, with the ejector system augmentation ratio, resultant force inclination, and resultant force displacement being determined from test rig load cell data. The ejector system augmentation ratio is defined as the ratio of the lifting capability of the ejector and the thrust at the manifold nozzle exits. Test data for ejector Configuration I are presented in Figures 53, 54, and 55 for its optimum ejector exit door angle of 4 degrees outward from the vertical. With both engines operating at an EPR of 2.28, this configuration had an ejector system augmentation ratio of 1.300. Additional data for this configuration are presented in Figure 57.

CONFIGURATION II

The best ejector performance with the D manifold was obtained with the arrangement referred to here as Configuration II. This configuration is defined in Figures 42, 43, 44, and 47. Test data for Configuration II are presented in Figures 56 and 57, and a comparison with Configuration I is made in the latter figure. At an EPR of 2.28, Configuration II had an ejector system augmentation ratio of 1.350, as compared to Configuration I, which had a ratio of 1.300.

LIFT PERFORMANCE - CONFIGURATIONS I AND II

These augmentation ratios, which were based on the D manifold thrust of 5,650 pounds, gave an ejector lift capability of 7,350 pounds and 7,630 pounds for ejector Configurations I and II, respectively, at an EPR of 2.28. To obtain the aircraft lift capability, the pitch reaction control contribution had to be added to the above values. This contribution was 450 pounds when both engines were operating at an EPR of 2.28. Hence, the aircraft lift capability was 7,800 pounds for Configuration I and 8,080 pounds for Configuration II.

F MANIFOLD

As a part of the lift improvement program, an attempt was made to fabricate a slot-type manifold using brazing techniques in order to improve the external surface smoothness of the manifold and thereby increase the ejector inlet pressure recovery factor. Brazing techniques did not prove to be satisfactory, and the manifold was assembled by welding. However, the external surface finish of this manifold, the F manifold, was somewhat smoother than that of the D manifold. A heavier gauge material was used to construct the F manifold than was used to construct the D manifold. This resulted in the F manifold's weighing approximately 80 pounds more than the D manifold, although the F manifold was considered to be more reliable. The F manifold is shown in Figures 33, 34, and 35.

The bare F manifold was then installed on the test rig, and its thrust was determined in the same manner as the thrust for the D manifold. Test data for the F manifold are presented in Figures 58, 59, and 60. The thrust at the nozzle exits of the F manifold was found to be 5,455 pounds at an EPR of 2.28, compared to a thrust of 5,650 pounds for the D manifold. The total exit area of the nozzles was measured for each manifold, and the area of the F manifold nozzles was found to be approximately 3 percent less than that of the D manifold nozzles. Therefore, the lower thrust level of the F manifold was attributed to a slight mismatch between the total nozzle exit area and the total engine exhaust exit area. The manifold did not have exit area trimming tabs to permit the correction of a mismatch such as this. Because of this area mismatch and resultant thrust difference, all comparative performance data are presented in terms of thrust augmentation ratios based on a primary nozzle thrust of 5,650 pounds for the D manifold and 5,455 pounds for the F manifold.

CONFIGURATION III

The F manifold was combined with the last ejector configuration that was flight tested, referred to here as Configuration III, and tested in the same manner as the previous configurations. Configuration III is defined in Figures 42, 43, 44, and 45 and, except for the different manifolds, is exactly the same as Configuration I. Test data for Configuration III are presented in Figures 61 and 62, and a comparison with Configuration I is made in the latter figure. At an EPR of 2.28, Configuration III had an ejector system thrust augmentation ratio of 1.348, as compared to Configuration I, which had a ratio of 1.300.

CONFIGURATION IV

The configuration referred to here as Configuration IV had the best performance of any ejector configuration that could be installed on the XV-4A aircraft. Configuration IV is defined in Figures 42, 43, 45, and 48. Test data for this configuration are presented in Figures 63 and 65. Configuration IV had an ejector system augmentation ratio of 1.450 at an EPR of 2.28.

CONFIGURATION V

The best performance of any ejector configuration tested was obtained from the configuration referred to here as Configuration V. This configuration, which is shown on the test rig in Figures 39, 40, and 41, had a bell-mouth-type ejector inlet that could not be installed on the XV-4A aircraft. Configuration V is defined in Figures 42, 43, 45, and 49. Test data for this configuration are presented in Figures 64 and 65, and a comparison with Configurations III and IV is made in the latter figure. Configuration V had an ejector system augmentation ratio of 1.480 at an EPR of 2.28.

LIFT PERFORMANCE - CONFIGURATIONS III, IV, AND V

The augmentation ratios for Configurations III, IV, and V were based on the F manifold thrust of 5,455 pounds. This gave an ejector lift capability of 7,360 pounds, 7,920 pounds, and 8,080 pounds for Configurations III, IV, and V, respectively, at an EPR of 2.28. Including the pitch reaction control contribution of 450 pounds, the aircraft lift capability was 7,810 pounds for Configuration III, 8,370 pounds for Configuration IV, and 8,530 pounds for Configuration V.

SYSTEM LOSSES

Total pressure loss performance test data for one engine with the F manifold are presented in Figure 67. These data indicate that the total pressure loss between the engine exit and the manifold nozzle exits averaged 8.66 percent for an EPR of 2.20.

Total temperature loss performance test data for one engine with the F manifold are presented in Figure 68. These data indicate that the total temperature loss between the engine exit and the manifold nozzle exits averaged 4.69 percent for an exhaust gas temperature (EGT) of 925° F.

REMARKS

In general, the VTOL lift improvement program results indicate that the jet ejector thrust was quite sensitive to the ejector inlet pressure losses and the ejector exit geometry as well as the nozzle geometry.

MISCELLANEOUS TESTS

SMALL-SCALE WIND TUNNEL TESTS

Wind tunnel tests of a 0.18 scale model of the XV-4A were conducted in the Chance Vought 7-by-10-foot wind tunnel to determine the aerodynamic characteristics that could be expected from the actual airplane during VTOL flight, transition, and CTOL flight. The major portions of the tests were conducted in the transition regime with various levels of ejector thrust.

With several differences, the ejector powered model shown in Figure 69 was a scale model of the XV-4A aircraft (wing span of 4.5 feet and length of 5.9 feet). The model wing was geometrically similar to the aircraft wing. However, the flap span of the model was 8.5 percent of the wing span less than that of the aircraft, and the aileron span of the model was 8.5 percent of the wing span more than that of the aircraft. The model engine nacelles were longer and had a slightly smaller frontal area than scale nacelles would have had. The model ejector inlet doors were not of the same configuration as those later used on the aircraft. The model differed from the aircraft empennage in that the horizontal tail of the model was movable, while the horizontal tail of the aircraft had a fixed incidence. Both the horizontal and vertical tails of the aircraft had slightly greater areas than those scaled from the model.

The model's two parallel ejectors had an area ratio of 14.5. The area ratio is the ejector exit area divided by the manifold nozzle exit area. The nozzle manifold consisted of a nozzle block which had a total of 80 individual nozzles arranged in 20 units of 4 nozzles each. Air was supplied from outside the wind tunnel to the manifold plenum chamber at a pressure of 300 pounds per square inch. Although the nozzle manifold was not an exact scale model of the airplane design configuration, scaled values of total system thrust, nozzle exit area, and the position and orientation of the nozzles with respect to the ejector chambers were maintained. As in the airplane, the center line of each of the model ejector sections was canted outboard at the bottom to an angle of 7 degrees and inclined rearward at an angle of 12 degrees.

For testing, the model was mounted in the wind tunnel on a conventional three-strut support system, as shown in Figure 69, with the rear strut also serving as a part of the system used to supply compressed air to the manifold. A trapeze-type ducting arrangement was used to eliminate forces and moments induced on the balance system by the external air supply system. Special calibration runs were made to determine the effects of the trapeze arrangement on the tunnel balance. Otherwise, conventional testing procedures were used, with six component data being recorded for each run. Corrections applied to the test data accounted for tunnel wall effects, support system effects, nacelle internal drag, and trapeze arrangement effects. For the ejector-powered runs, the tunnel wall effect corrections were applied to the aerodynamic portion of the total lift. The small-scale wind tunnel data are presented in Reference 1.

As a result of these tests, the airplane horizontal tail area was increased slightly, and a fixed incidence was incorporated for this surface. The vertical tail area was also increased slightly. In addition, the pitch reaction control power was increased by approximately 44 percent, and a blowing boundary layer control system was applied to the leading edge of the horizontal tail and elevator. The boundary layer control permitted the use of large elevator deflections during transitional flights.

An attempt was made to correlate the small-scale data for various thrust levels and airspeeds by the use of a thrust coefficient. This did not produce acceptable correlation; therefore, generalized dimensional equations were derived for all forces and moments. Since significant differences existed between the airplane and model horizontal tails, the longitudinal equations were derived from tail-off model data, with additional terms being added to account for the airplane horizontal tail. The lateral and directional equations were derived from tail-on model data, since it was estimated that the differences between the model and airplane empennage would have a negligible effect on these characteristics. Where applicable, terms were added to account for the landing gear, the engine inlet ram drag, and the thrust of the horizontal thrusting engine during Phase II flight. The addition of the reaction control terms to the generalized equations produced the predicted full-scale airplane data.

FULL-SCALE WIND TUNNEL TESTS

One of the XV-4A aircraft was tested in the National Aeronautics and Space Administration (NASA) Ames Research Center's 40-by-80-foot wind tunnel to investigate the VTOL flight, transition, and CTOL flight characteristics of the actual aircraft. Flight testing had been completed prior to the full-scale wind tunnel testing.

For testing, the aircraft was mounted in the tunnel on the conventional three-strut support system, as shown in Figure 70. The main landing gear was removed, and special fittings, which were attached to the main gear trunnions, were mated with and attached to the ball and socket joints at the upper ends of the main support struts. A special fuselage fitting was required to mate with the ball and socket joint of the rear support strut.

The ejection seat and most of the cockpit instrumentation were removed, and remotely controlled electric actuators were attached to the primary flight controls in the cockpit area. The aircraft control system was utilized to position the aerodynamic control surfaces and the reaction controls. The operation of all aircraft systems was remotely controlled. Fuel from an external source was supplied to each engine.

In the Phase I and Phase II flight regimes, major differences were found between the full-scale wind tunnel data and the aircraft characteristics that were predicted from the small-scale wind tunnel tests. The full-scale tests indicated a higher degree of stability about all axes than that predicted from the small-scale tests and also indicated less static lift capability than that observed during the flight tests.

In general, the full-scale data for the VTOL regime did not correlate well with data from the small-scale tests and the flight tests. Better correlation of data was obtained in the CTOL regime. Data from the full-scale wind tunnel tests are presented in Reference 2.

STRUCTURAL STATIC TESTS

The aileron, elevator, and rudder control systems were operationally tested at limit loads prior to the first aircraft flight. The flaps and landing gear were operationally tested when loaded to simulate the maximum air loads that these systems would encounter in flight.

A complete airframe static test was not performed. However, much of the airframe was tested in the process of testing the control systems.

FLUTTER AND VIBRATION TESTS AND PLACARD RESTRICTIONS

Ground vibration tests were conducted and, after minor modifications to the aircraft, the XV-4A was structurally cleared for operation at the altitudes and speeds shown in Figure 71.

The aircraft placard altitudes and speeds are shown in Figure 72.

Figure 1. XV-4A in Hovering Flight.

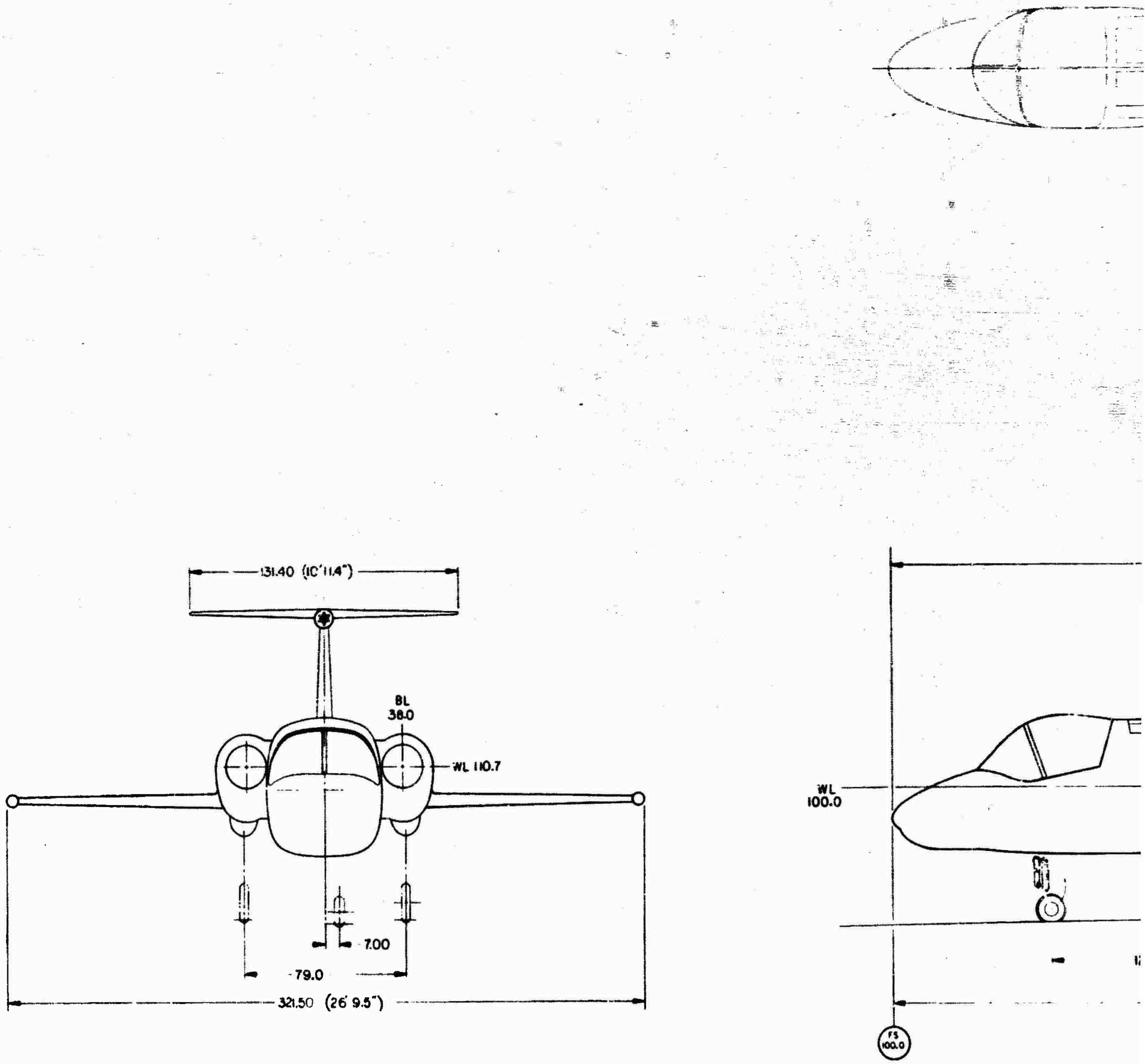
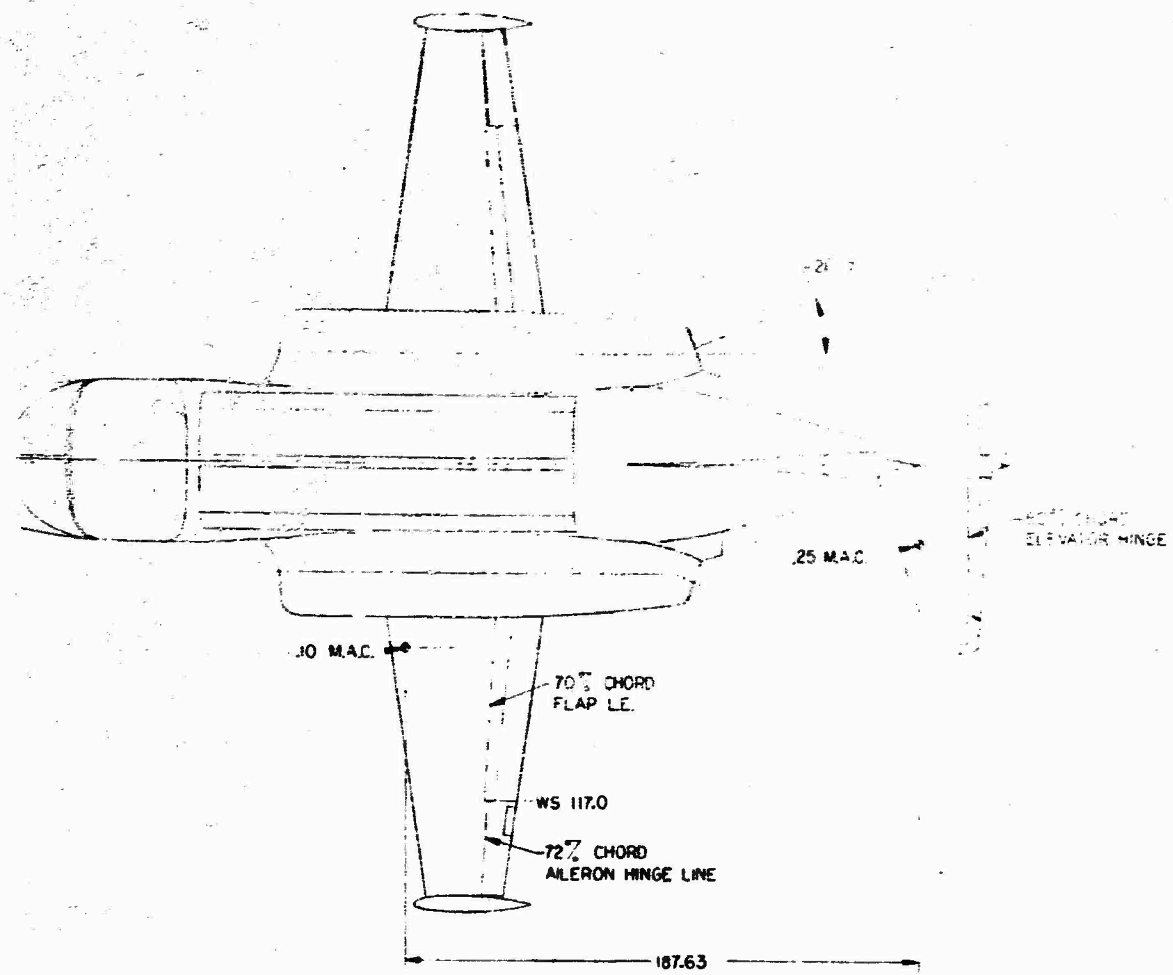
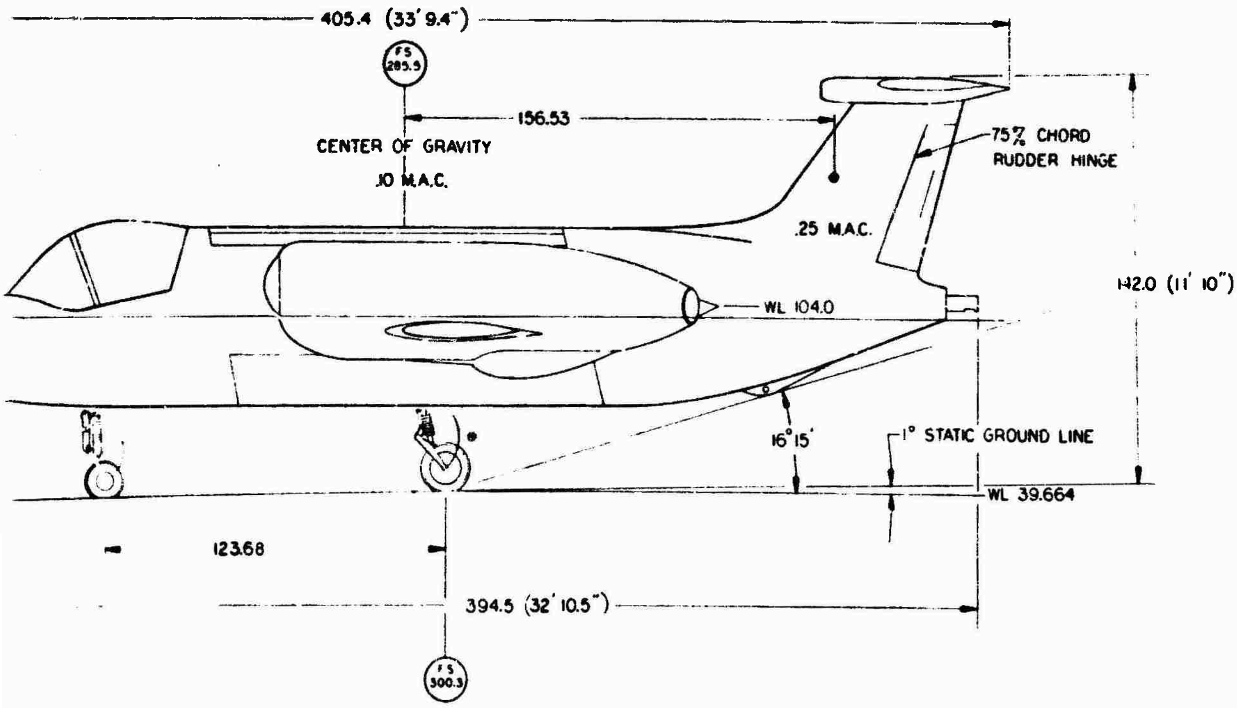


Figure 2. XV-4A General Arrangement.



ELEVATOR HINGE
.25 M.A.C.
.10 M.A.C.
70% CHORD
FLAP L.E.
WS 117.0
72% CHORD
AILERON HINGE LINE
187.63



405.4 (33' 9.4")
FS 285.5
156.53
CENTER OF GRAVITY
.10 M.A.C.
75% CHORD
RUDDER HINGE
.25 M.A.C.
142.0 (11' 10")
WL 104.0
16°15'
1° STATIC GROUND LINE
WL 39.664
123.68
394.5 (32' 10.5")
FS 300.3

B

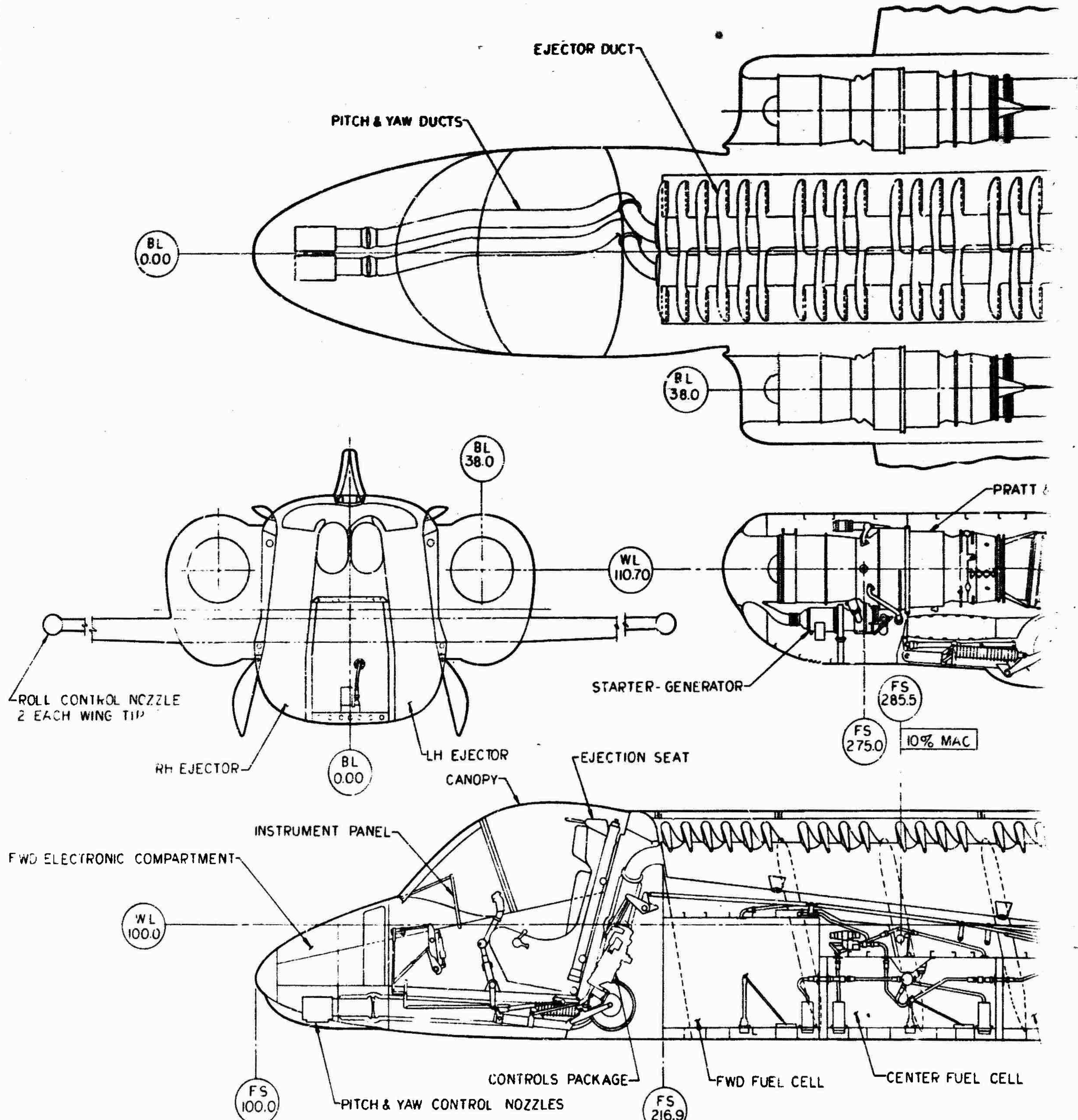
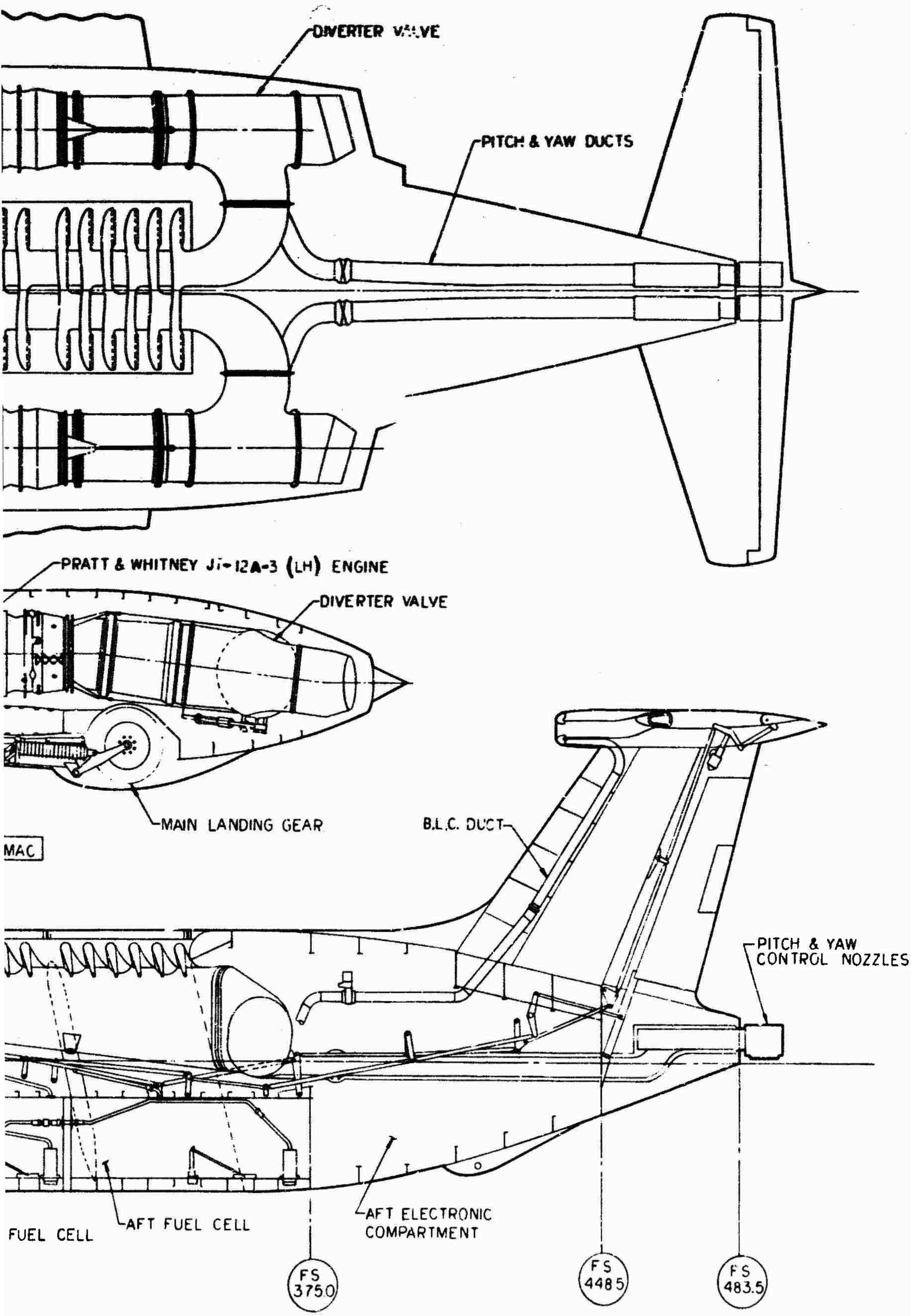


Figure 3. Inboard Profile.



DIVERTER VALVE
PITCH & YAW DUCTS
PRATT & WHITNEY J7-12A-3 (LH) ENGINE
DIVERTER VALVE
MAIN LANDING GEAR
MAC
B.L.C. DUCT
PITCH & YAW
CONTROL NOZZLES
FUEL CELL
AFT FUEL CELL
AFT ELECTRONIC
COMPARTMENT
FS 375.0
FS 448.5
FS 483.5

B

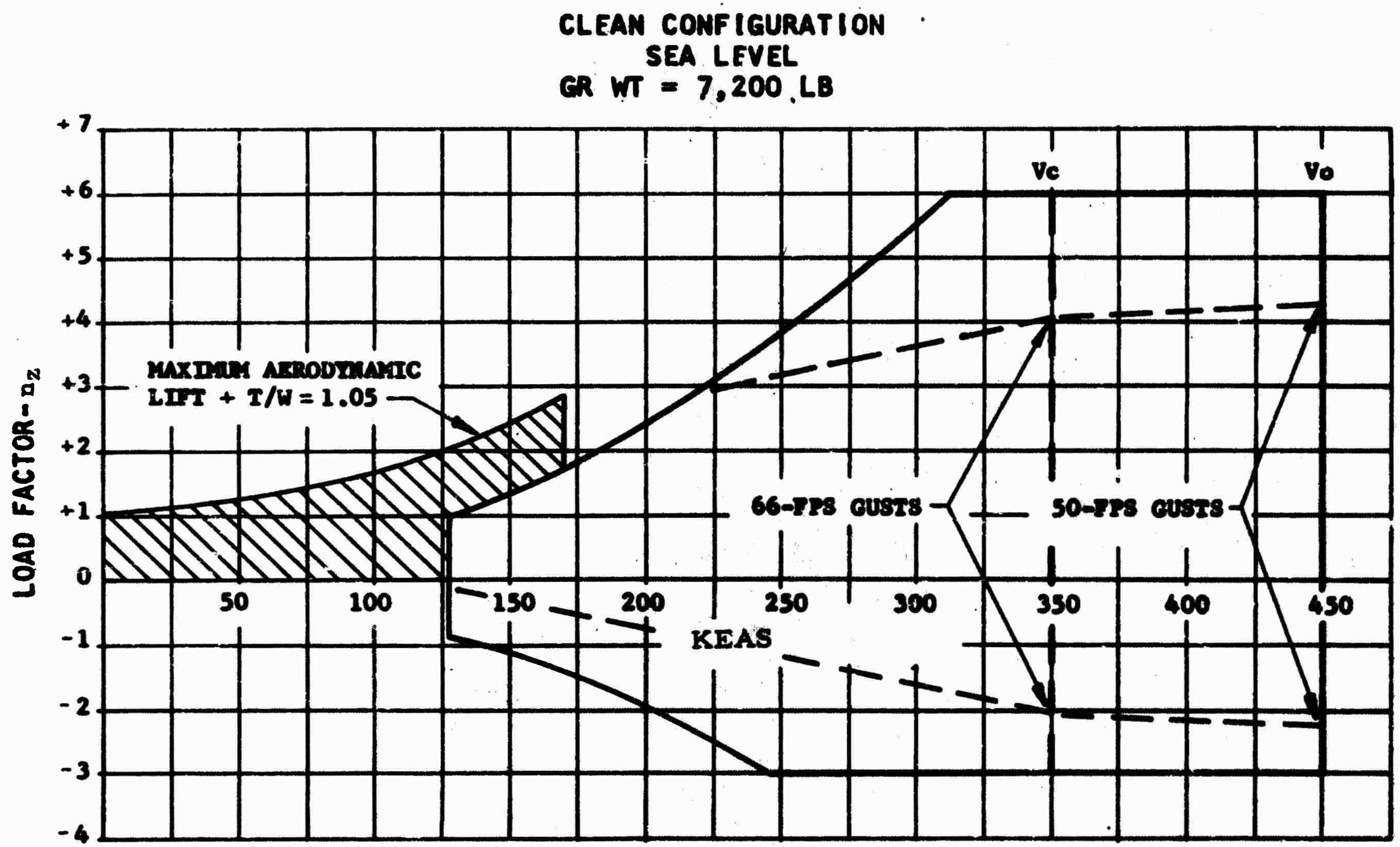


Figure 4. XV-4A Design V-n Diagram.

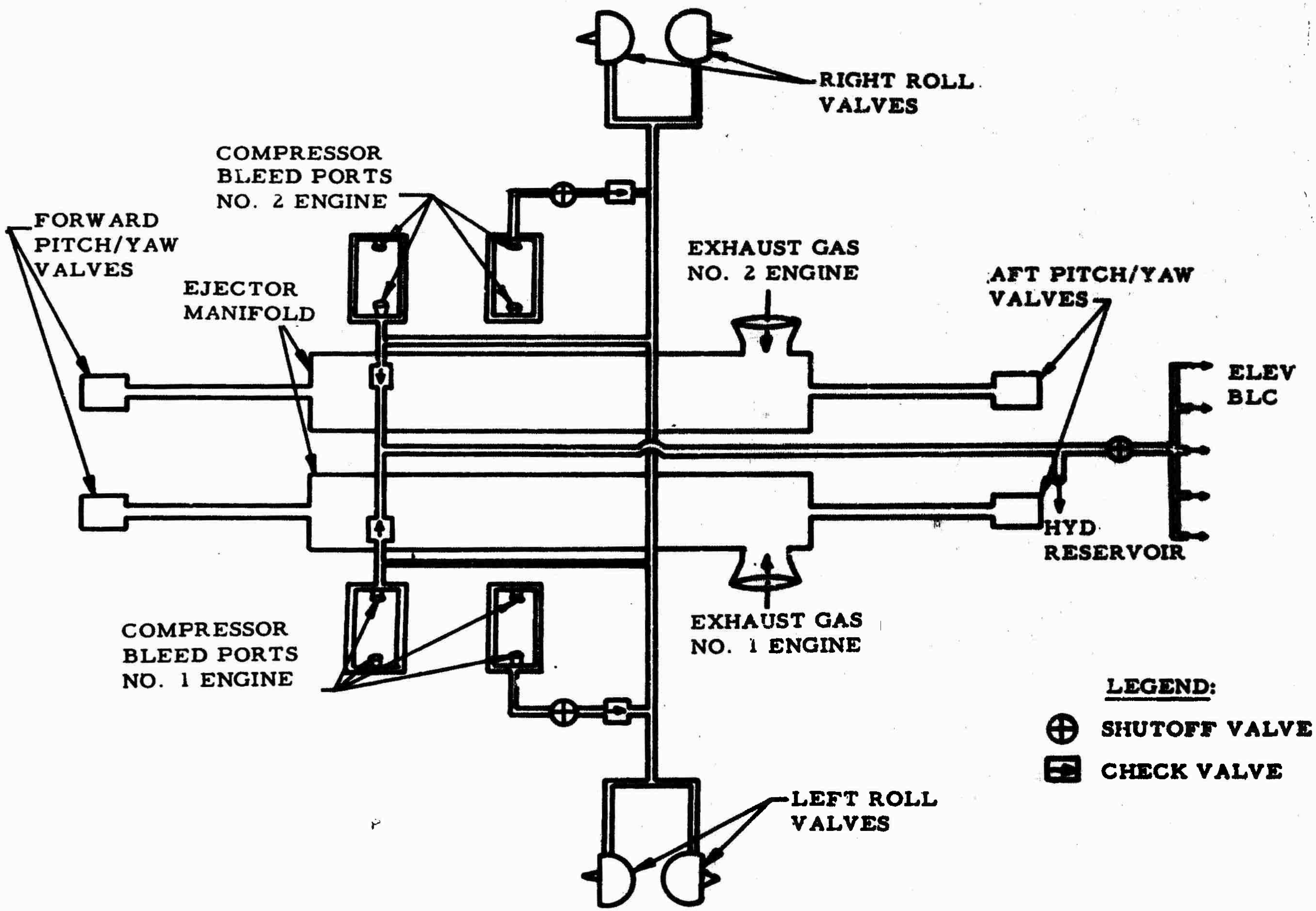


Figure 5. Pneumatic Power System.

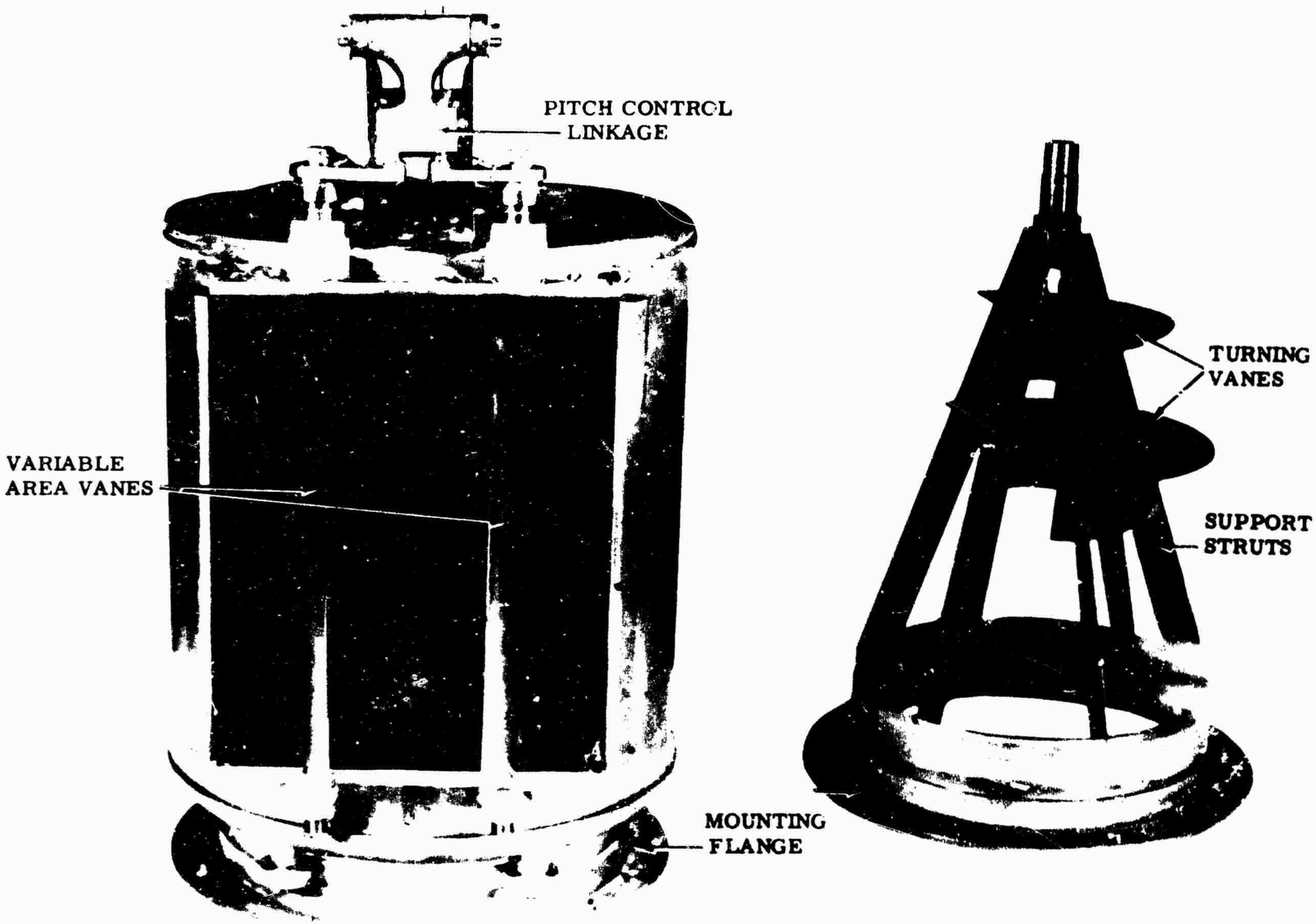


Figure 6. Pitch/Yaw Reaction Control Valve.

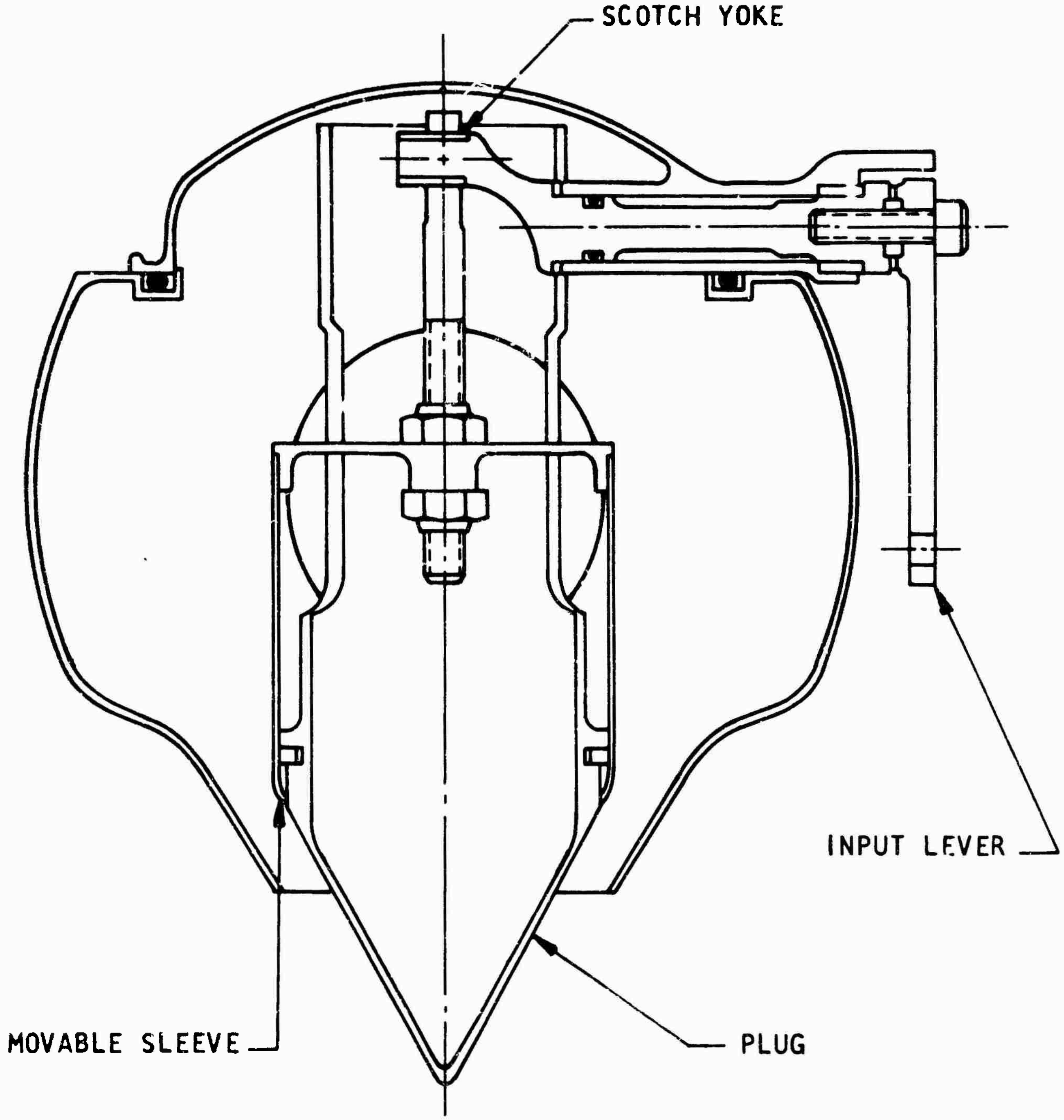


Figure 7. Roll Reaction Control Valve.

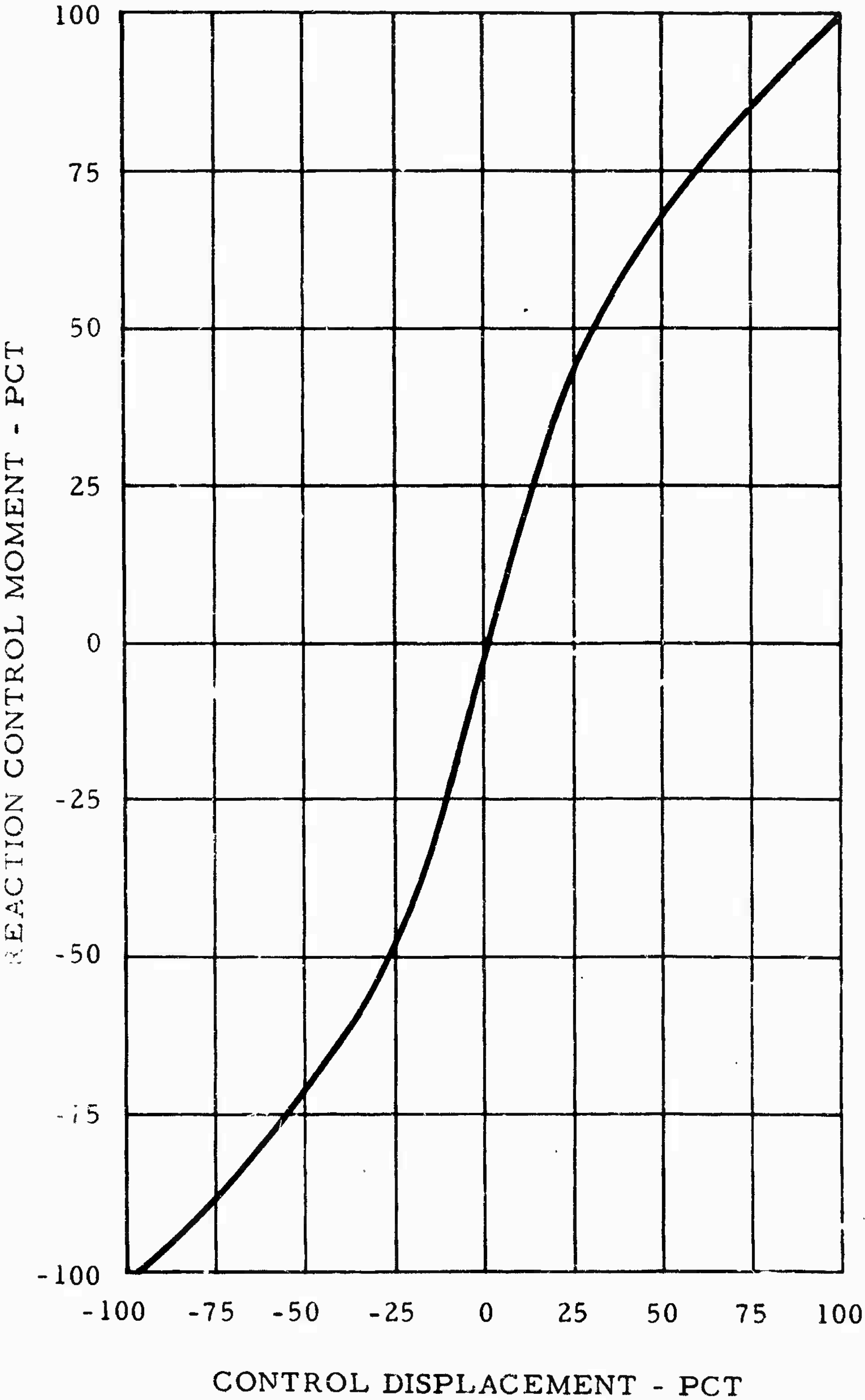


Figure 8. Reaction Control Moment Characteristics.

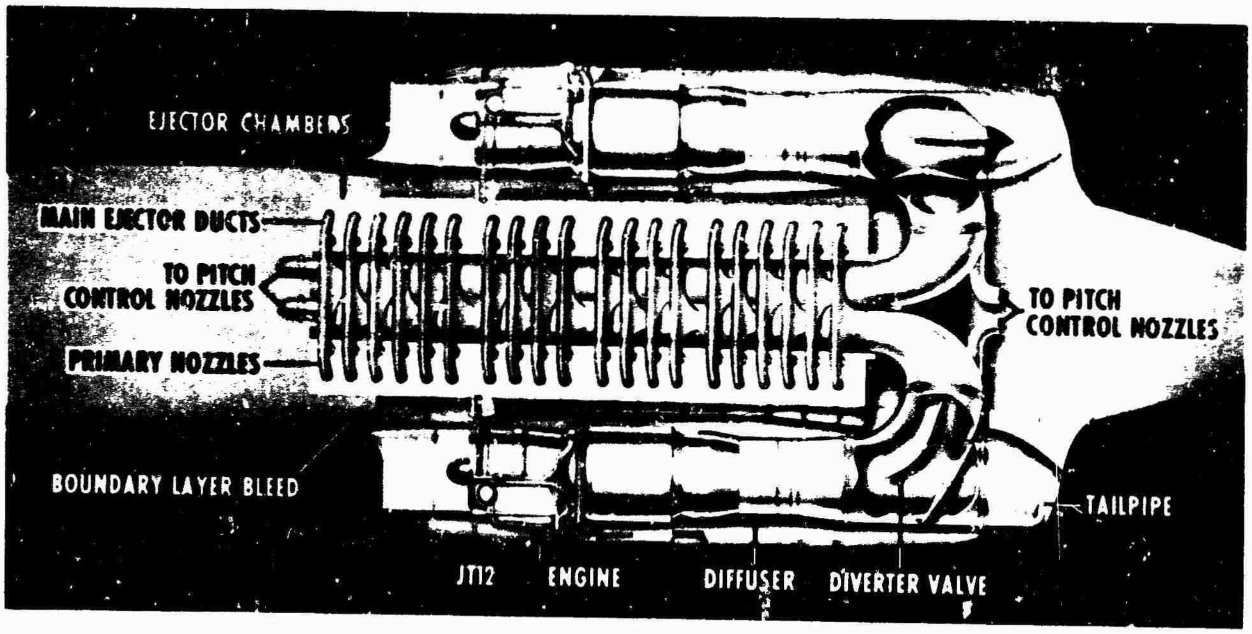


Figure 9. Propulsion System Schematic.

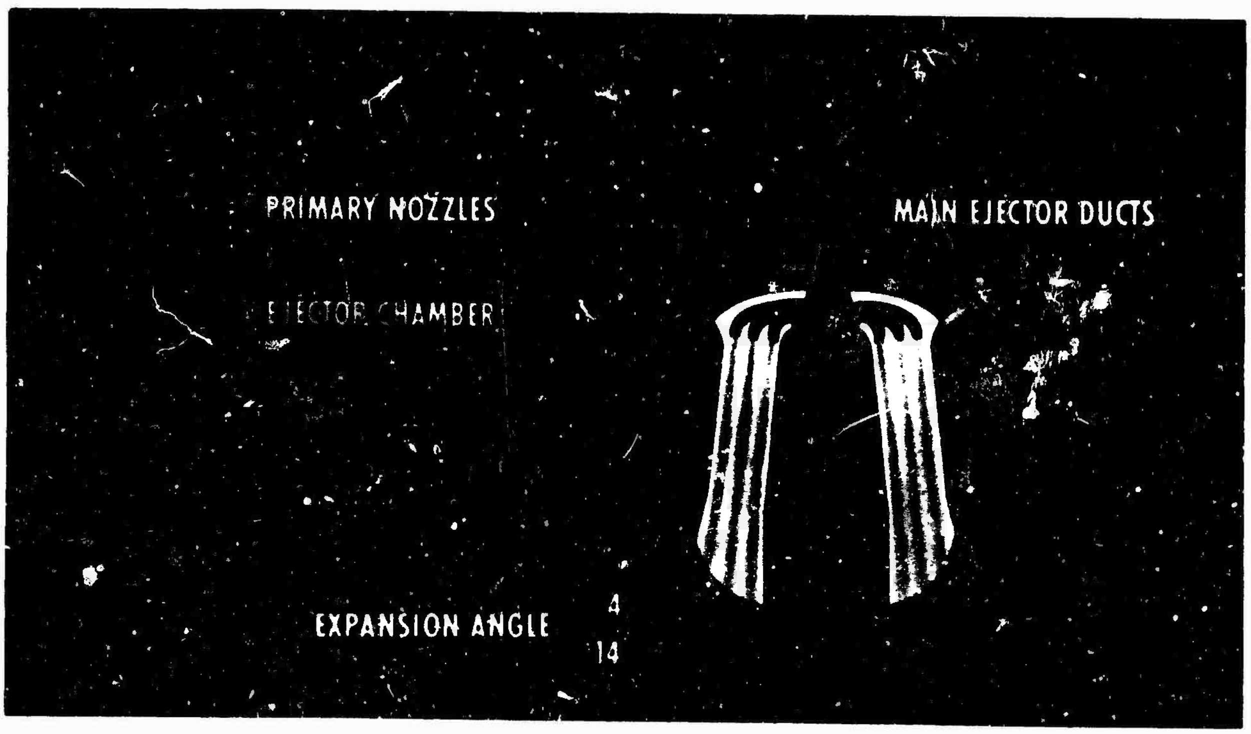


Figure 10. Ejector Schematic.

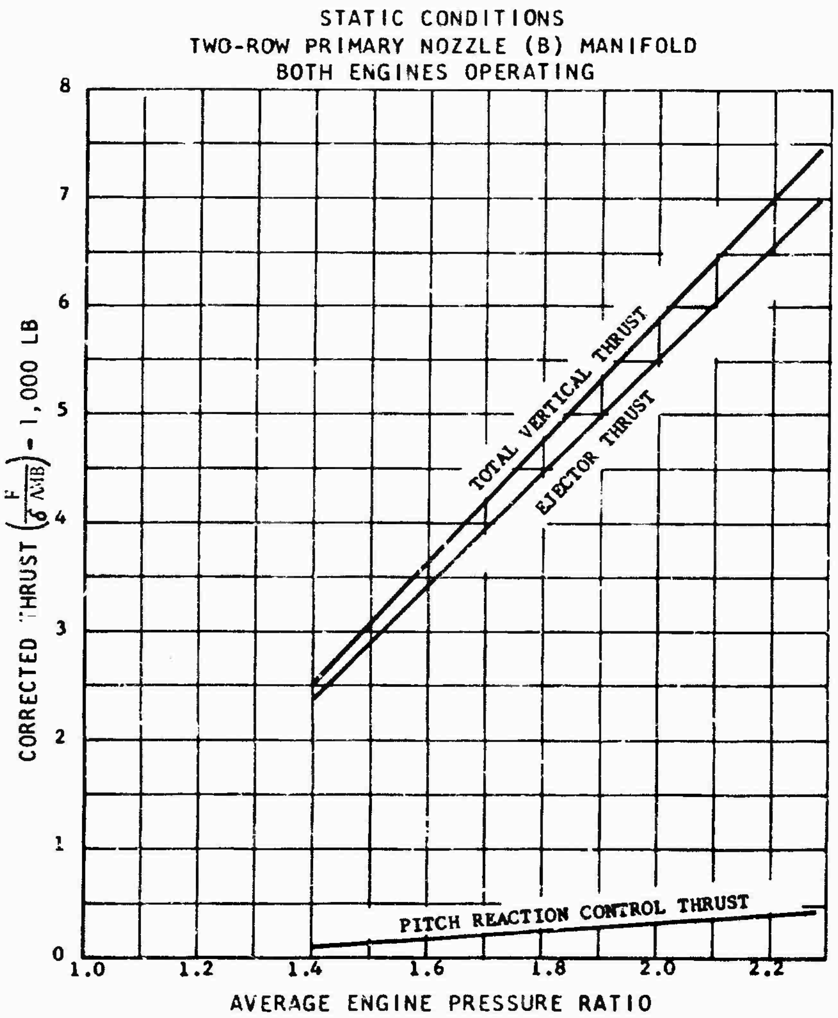


Figure 11. VTOL Propulsion System Performance; B Manifold.

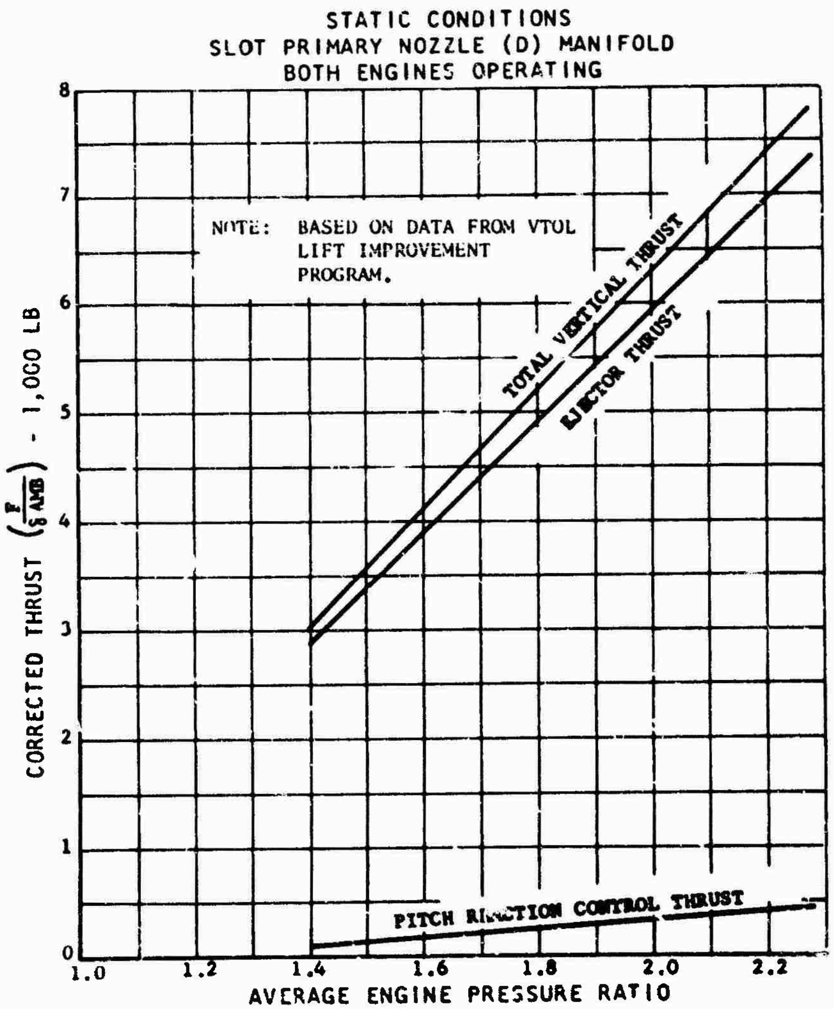


Figure 12. VTOL Propulsion System Performance; D Manifold.

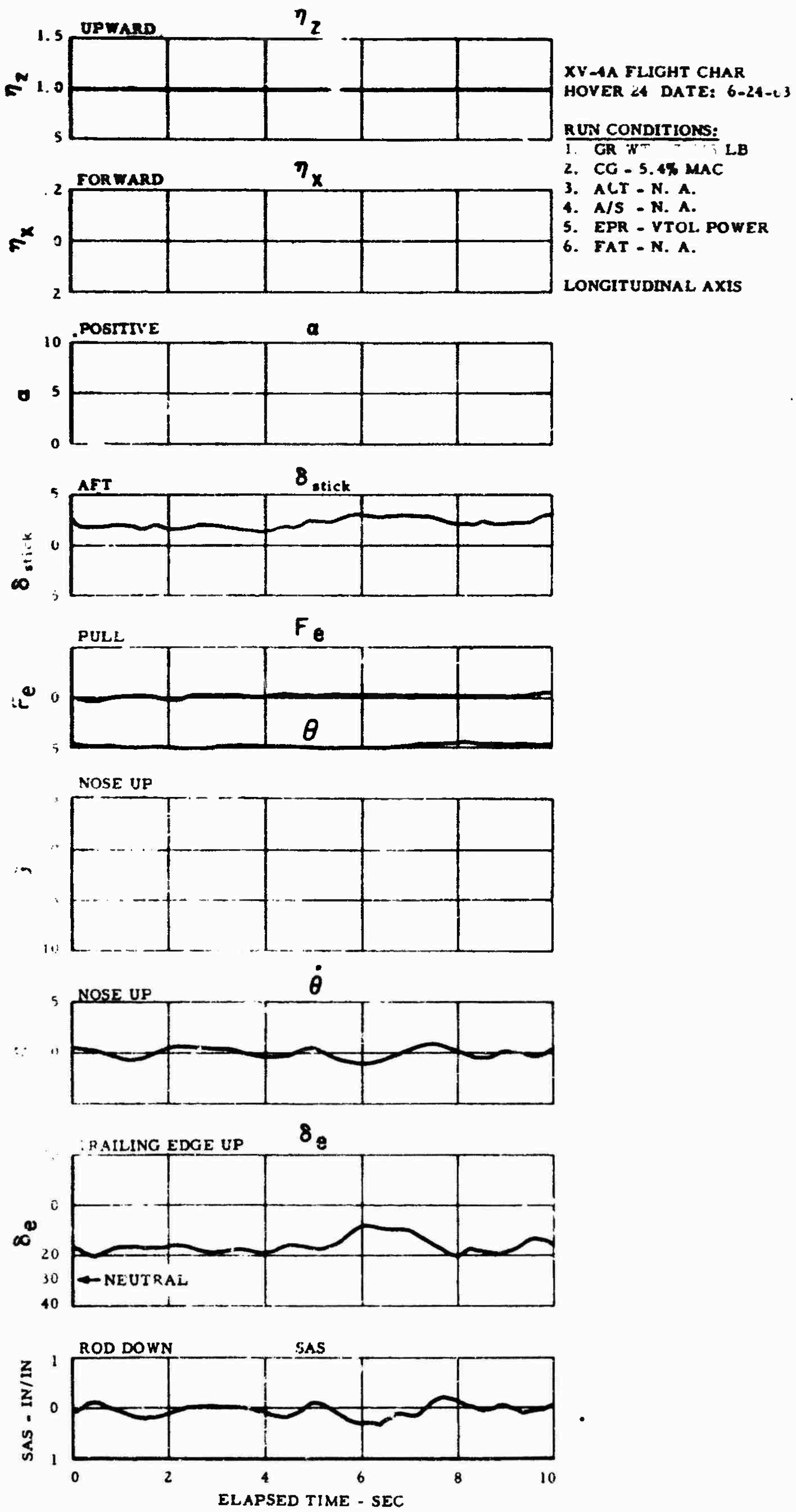


Figure 13A. Hover Flight Time History; Run 1.

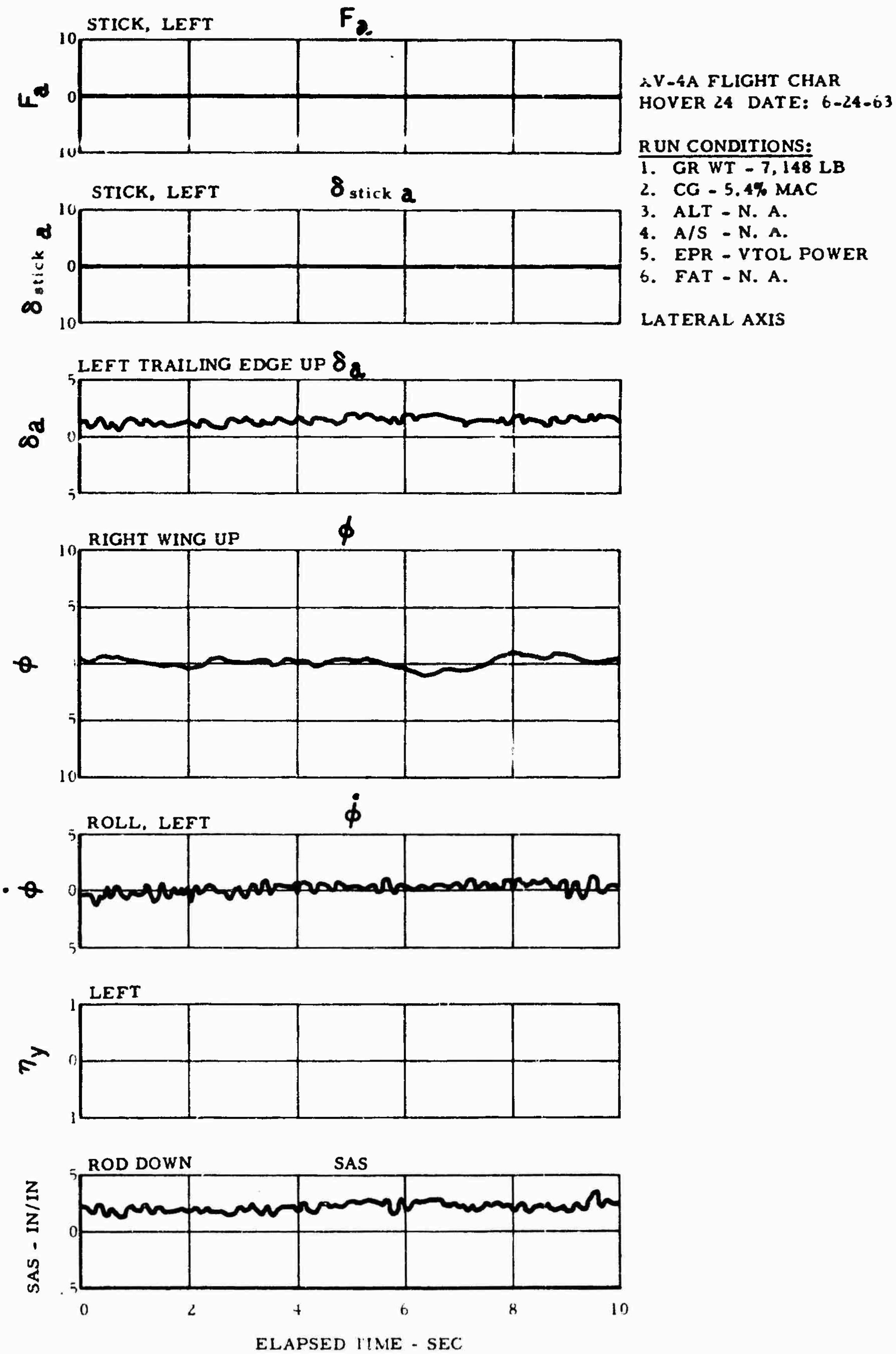
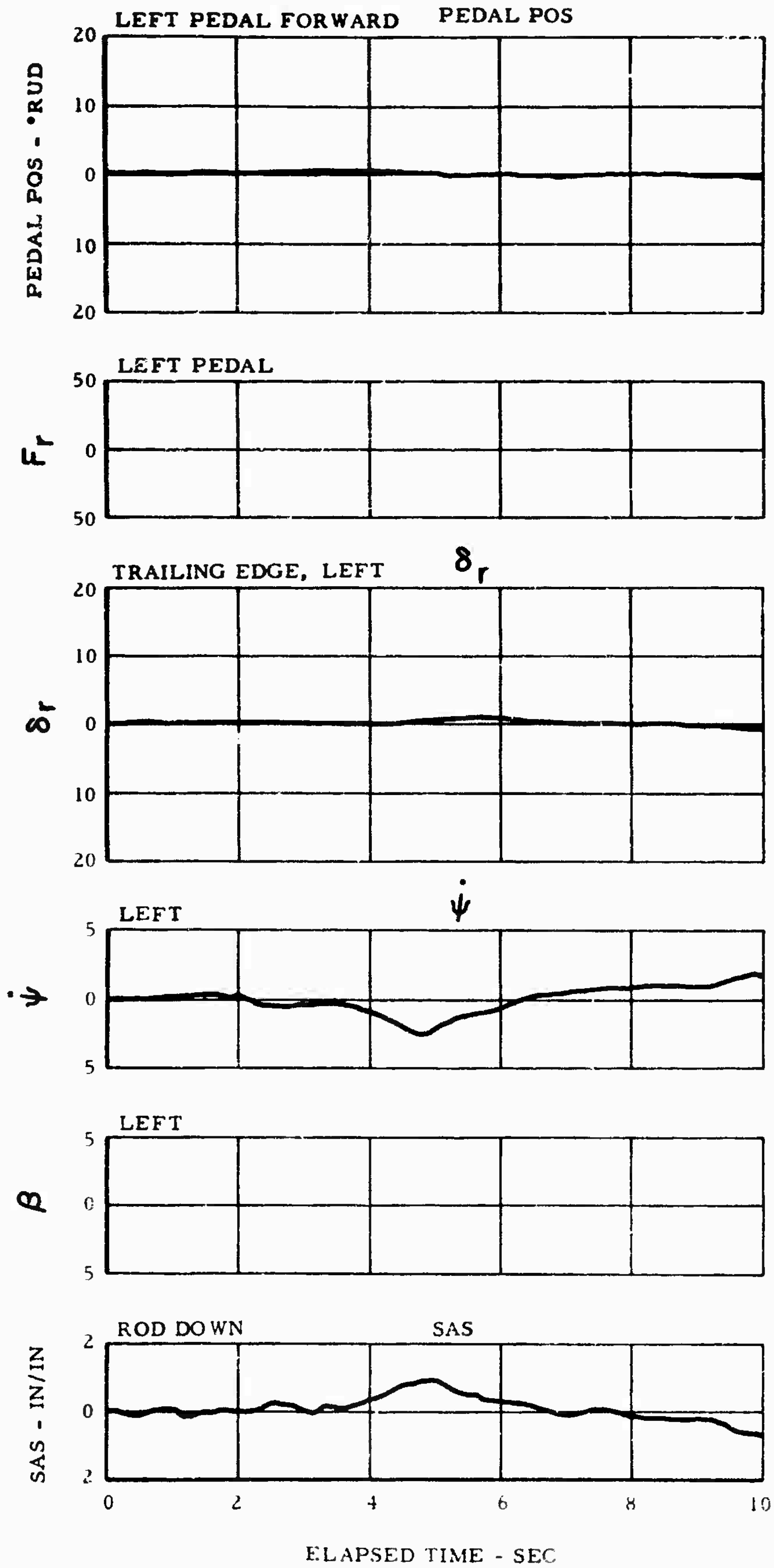


Figure 13B. Hover Flight Time History; Run 1.

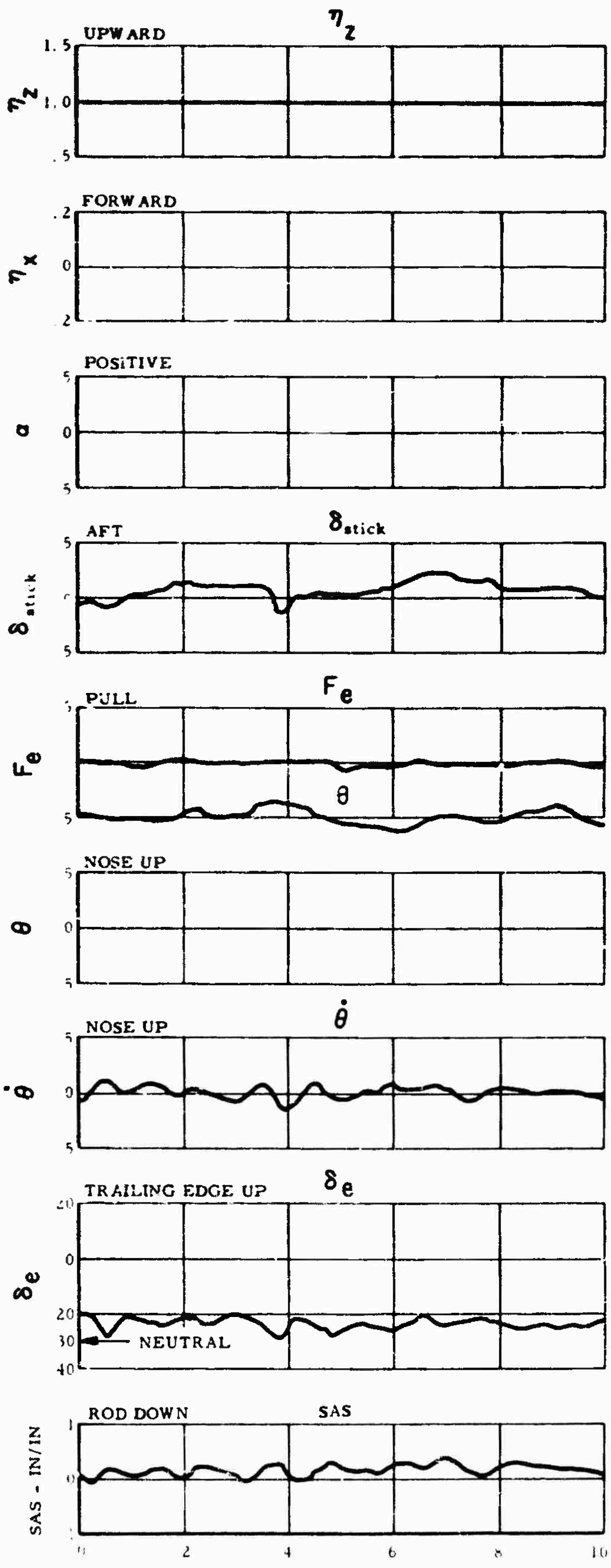


XV-4A FLIGHT CHAR
HOVER 24 DATE: 6-24-63

RUN CONDITIONS:
1. GR WT - 7,148 LB
2. CG - 5.4% MAC
3. ALT - N. A.
4. A/S - N. A.
5. EPR - VTOL POWER
6. FAT - N. A.

DIRECTIONAL AXIS

Figure 13C. Hover Flight Time History; Run 1.



XV-4A FLIGHT CHAR
HOVER 24 DATE: 6-24-64

RUN CONDITIONS:
1. GR WT - 7,148 LB
2. CG - 5.4% MAC
3. ALT - N. A.
4. A/S - N. A.
5. EPR - VTOL POWER
6. FAT - N. A.

LONGITUDINAL AXIS

Figure 14A. Hover Flight Time History; Run 2.

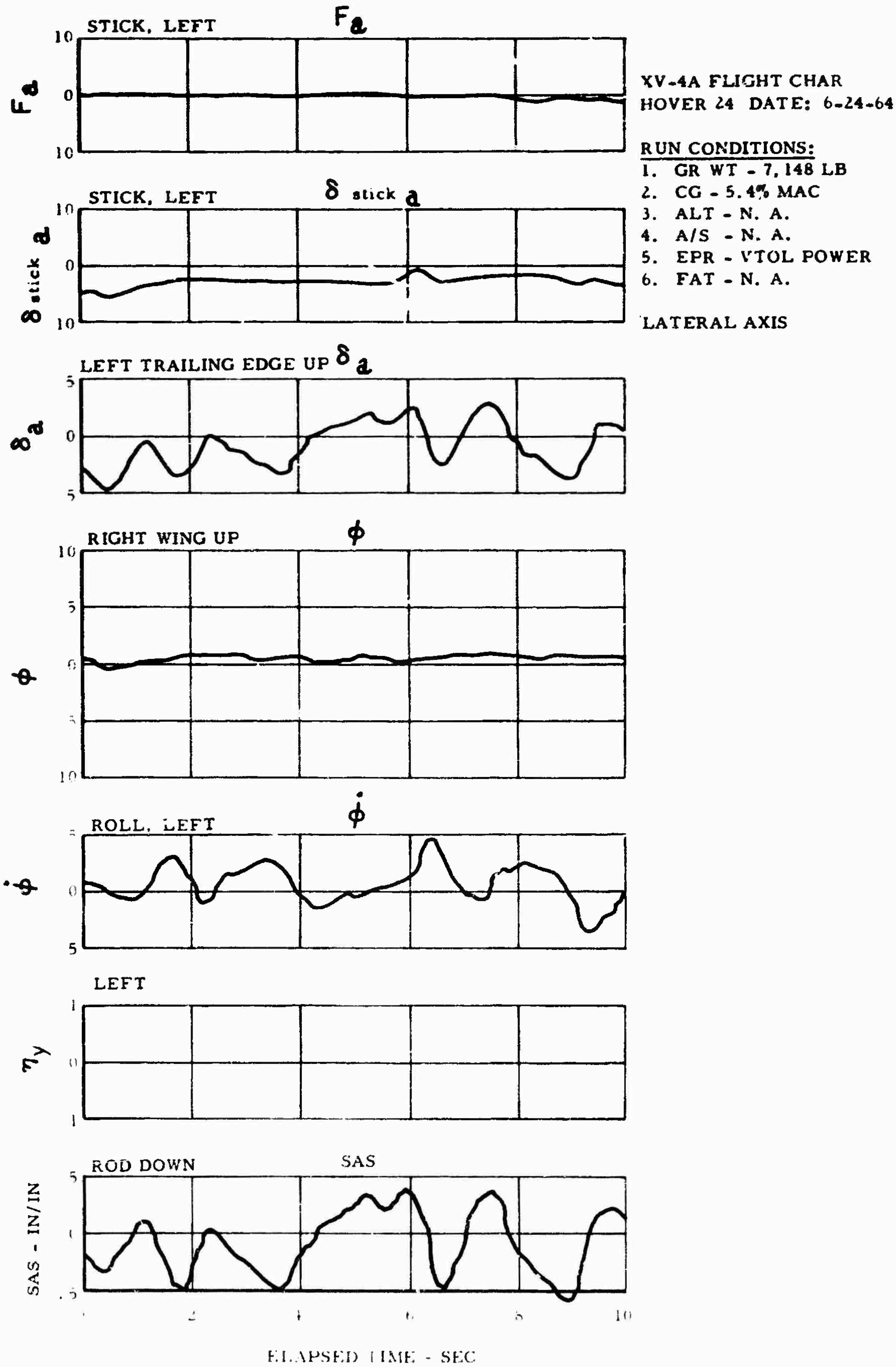


Figure 14B. Hover Flight Time History; Run 2.

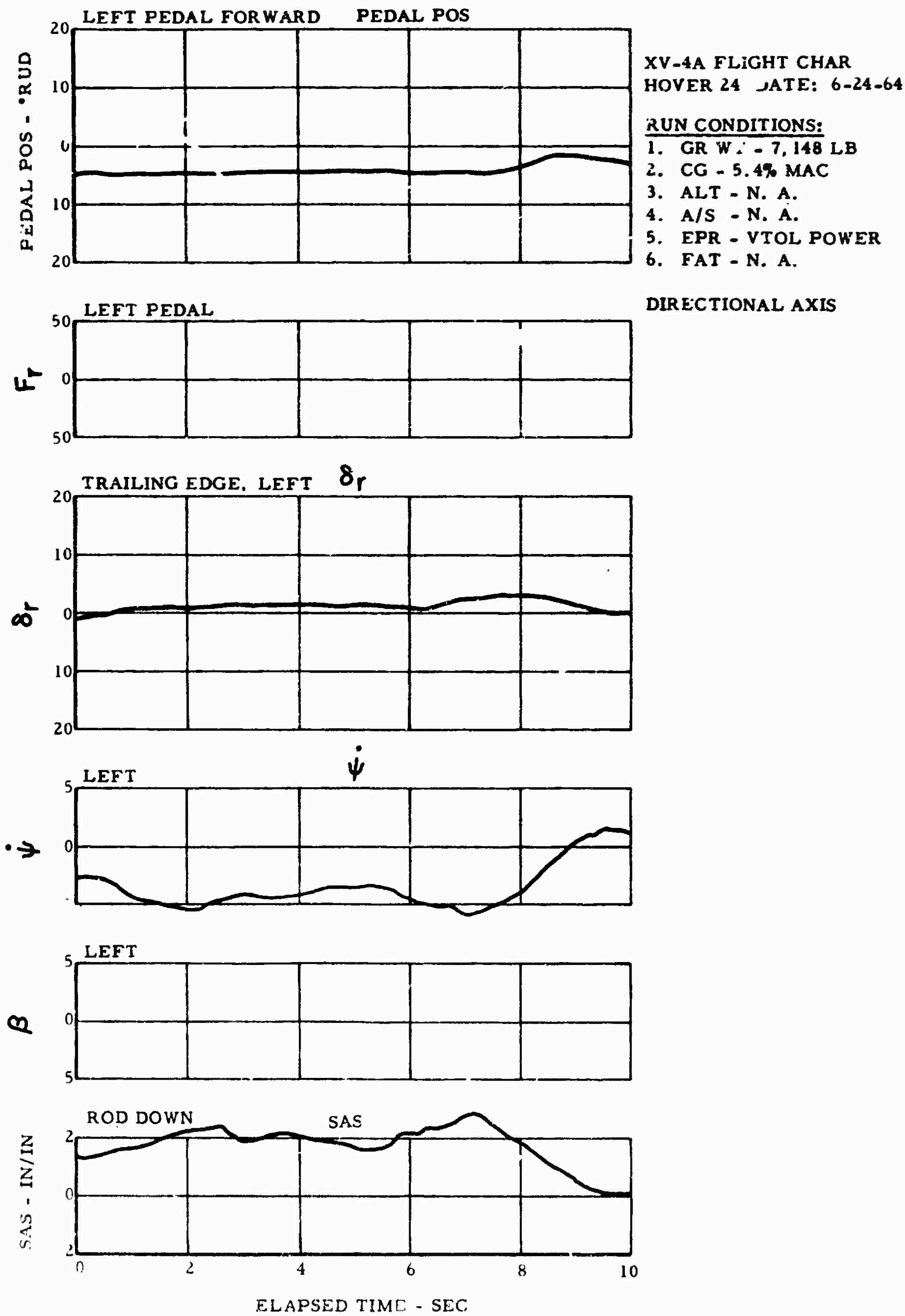


Figure 14C. Hover Flight Time History; Run 2.

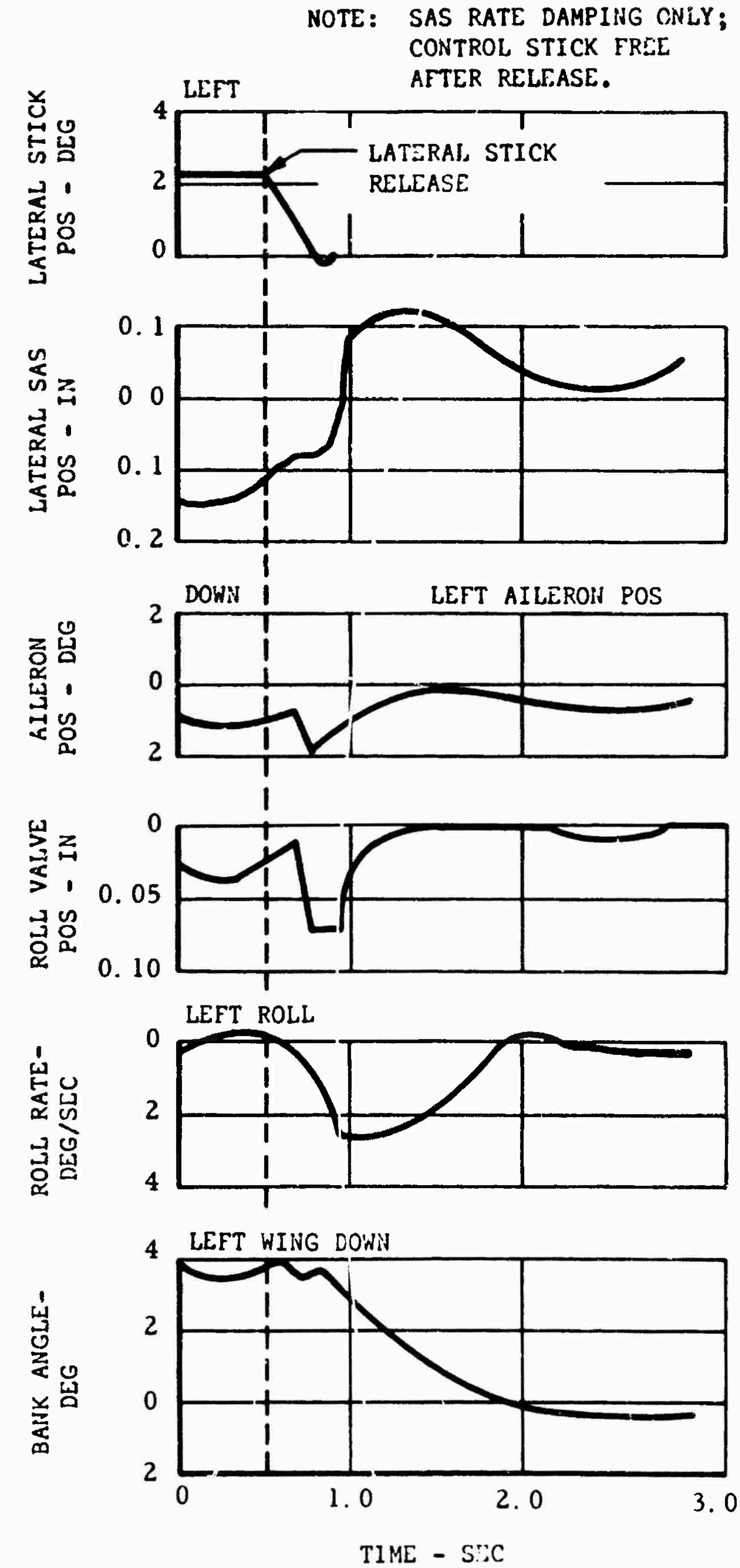


Figure 15. Time History of a Lateral Stick Release From Hover.

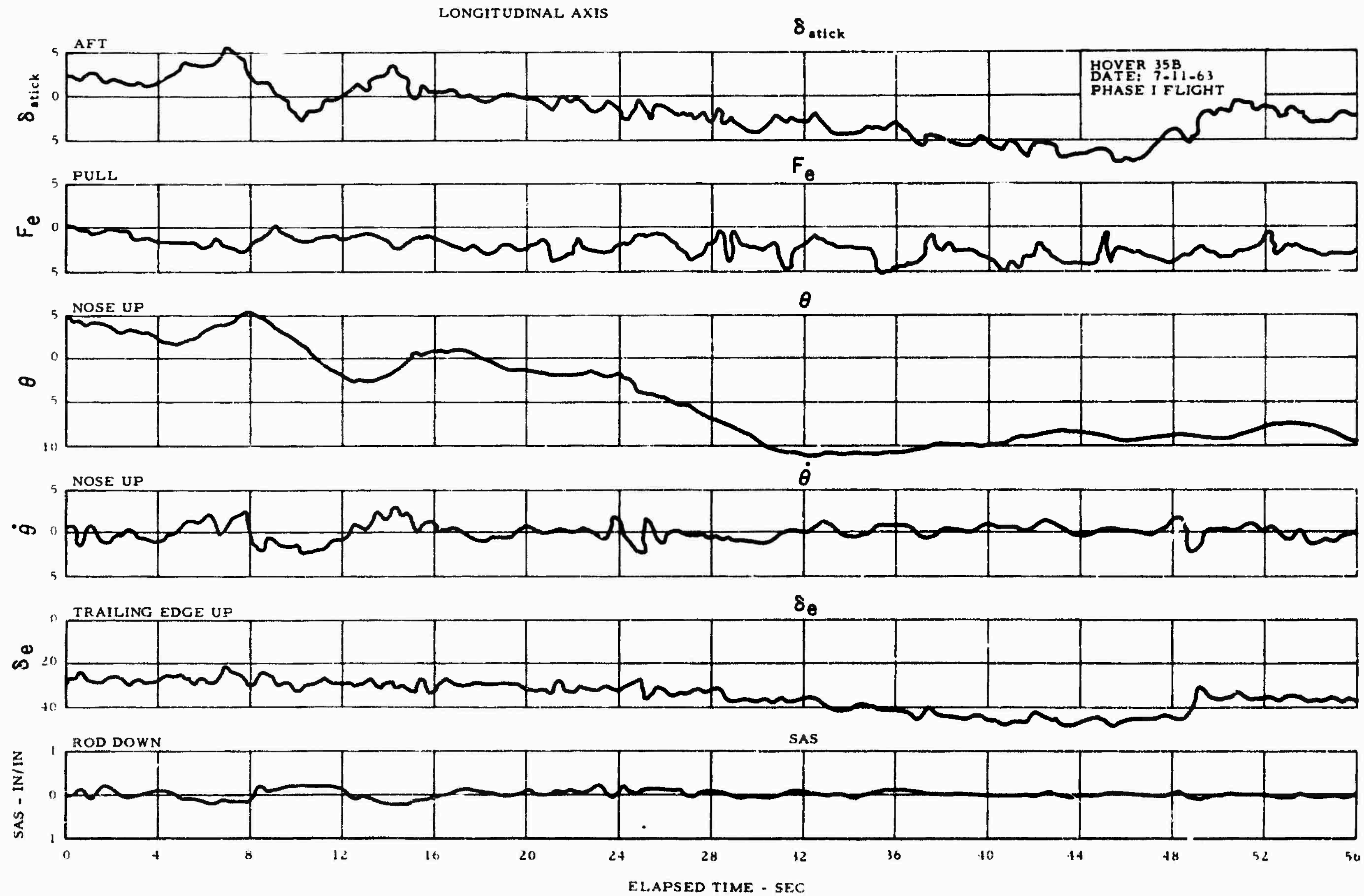


Figure 16A. Time History of a Phase I Translational Flight.

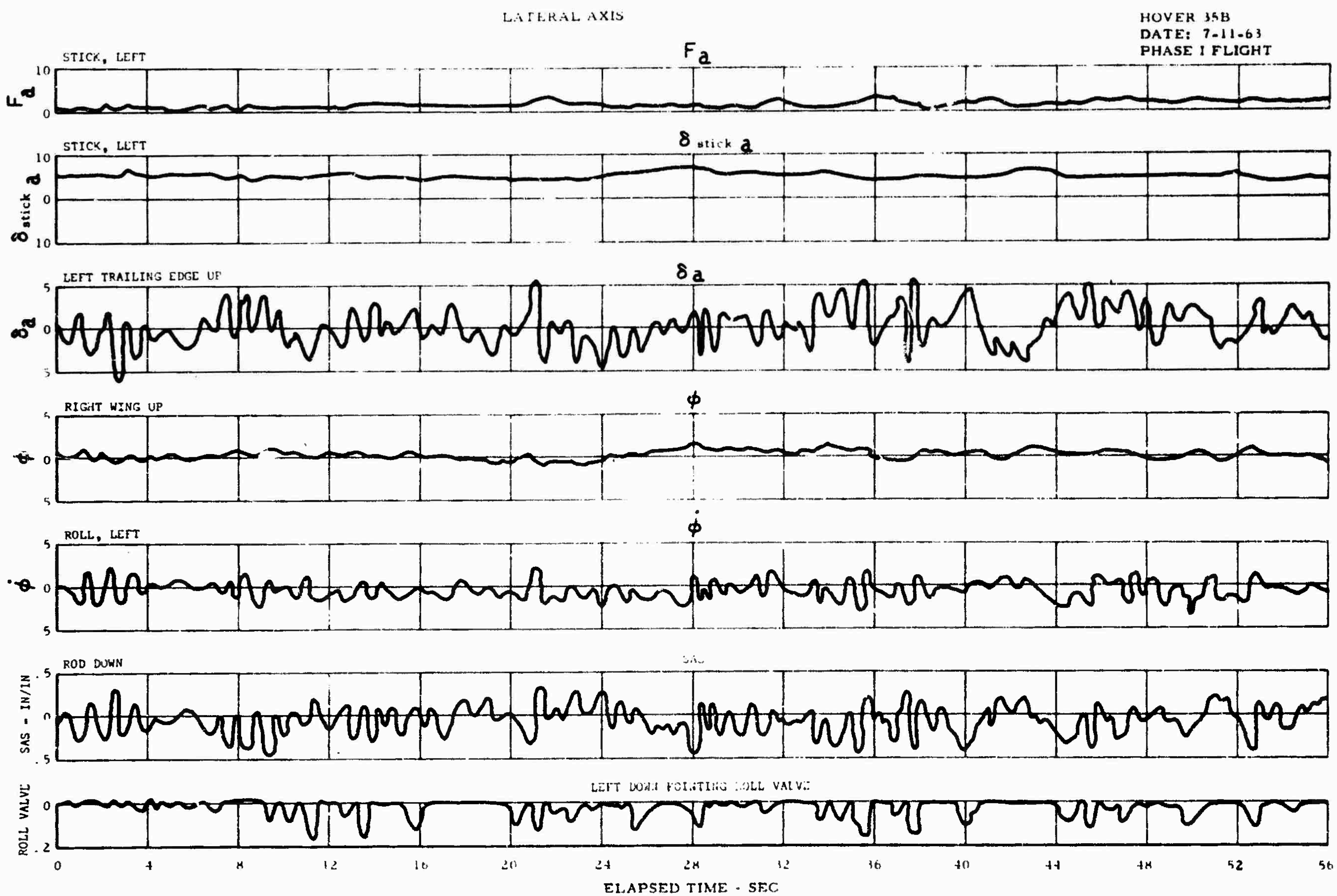


Figure 16B. Time History of a Phase I Translational Flight.

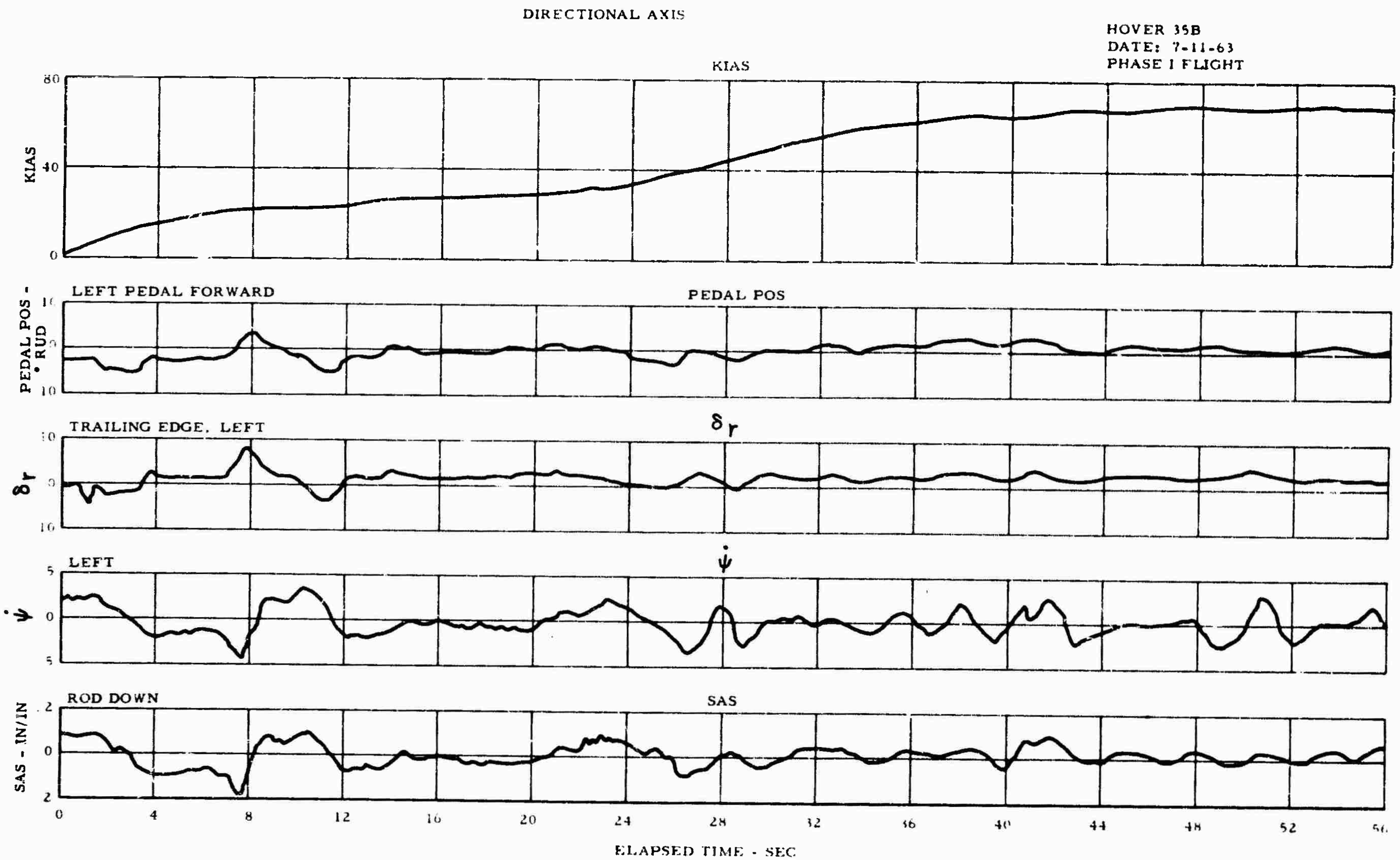


Figure 16C. Time History of a Phase I Translational Flight.

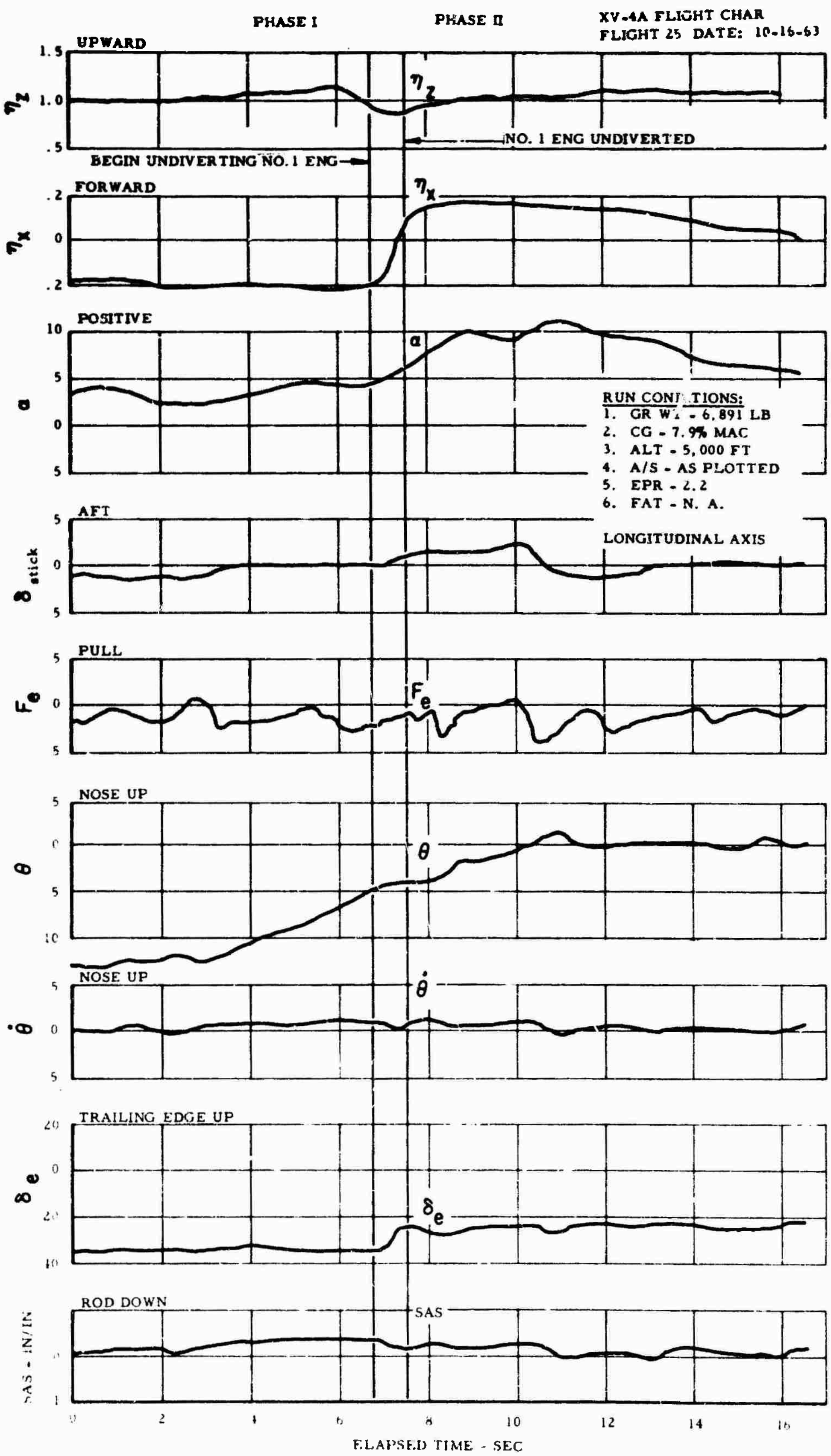


Figure 17A. Time History of an In-Flight Transition; Phase I to Phase II.

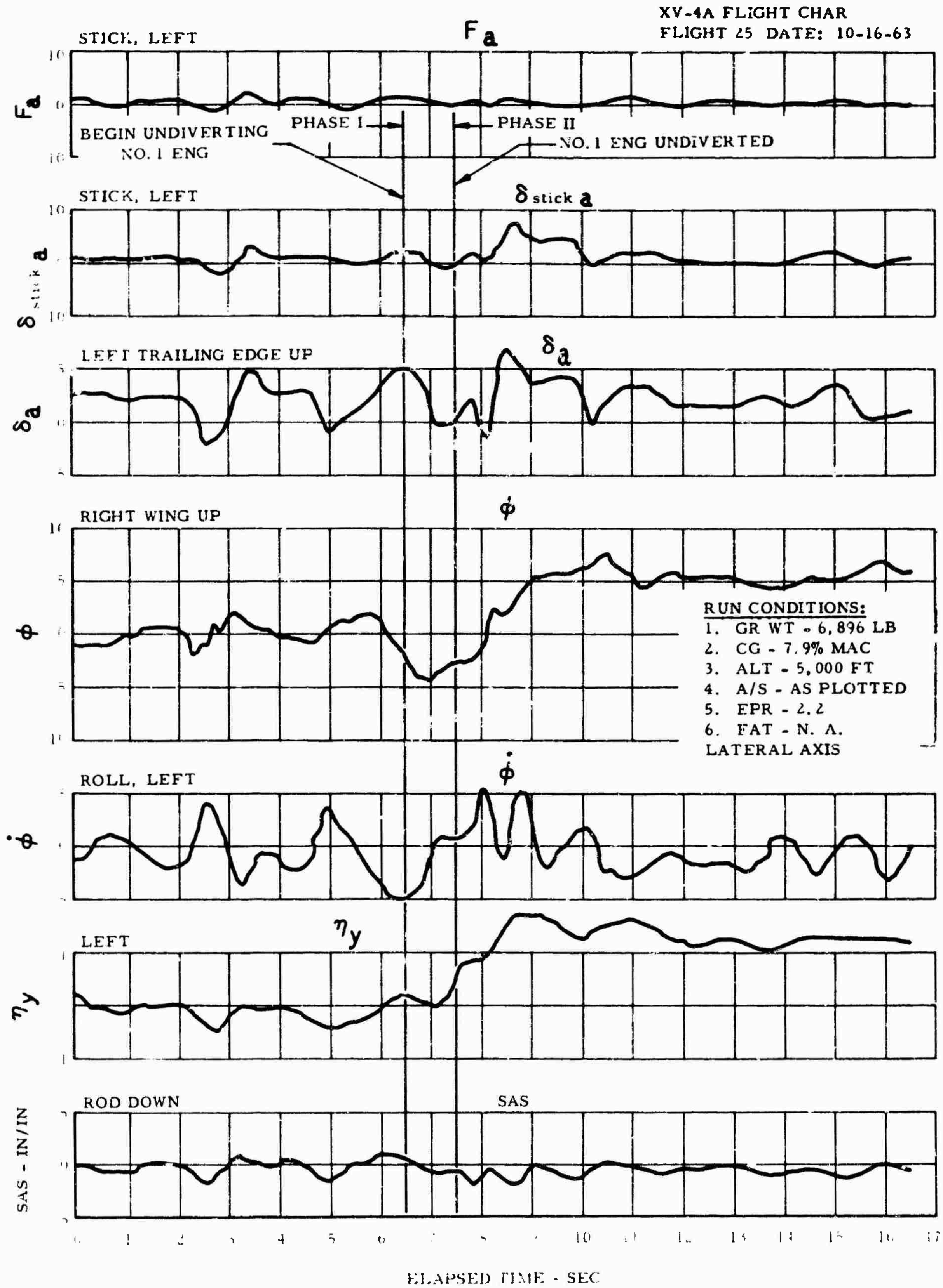


Figure 17B. Time History of an In-Flight Transition; Phase I to Phase II.

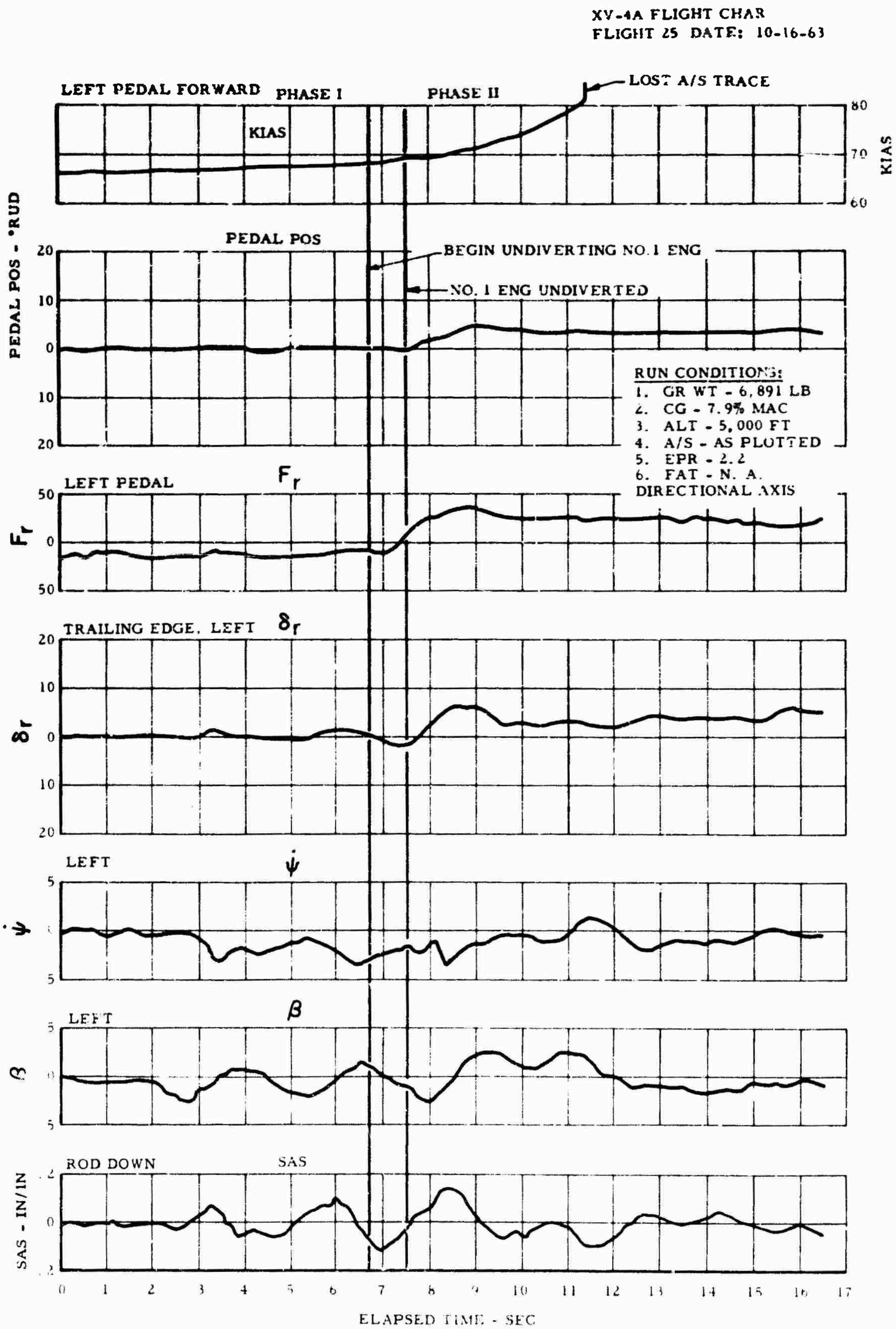


Figure 17C. Time History of an In-Flight Transition; Phase I to Phase II.

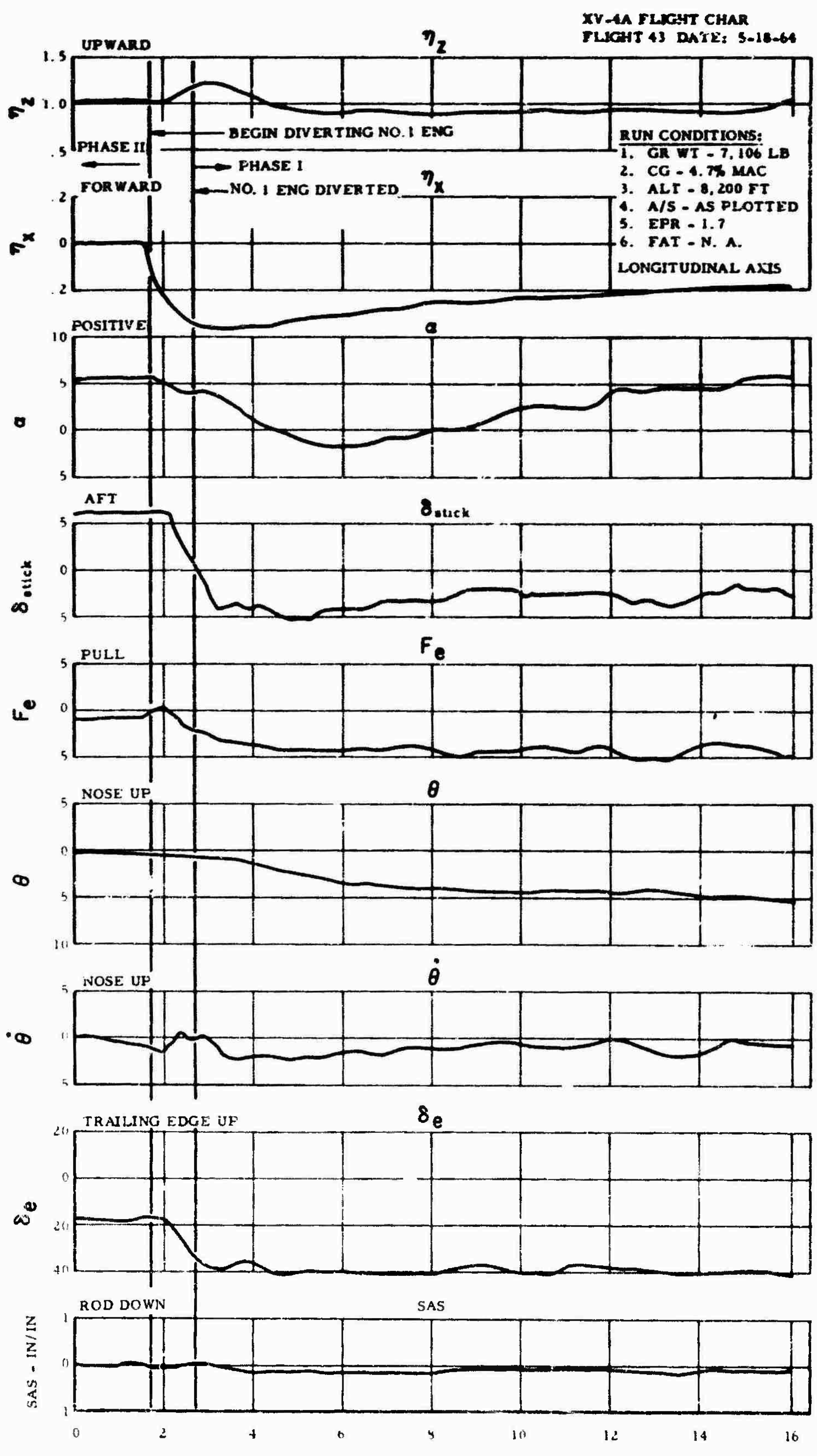


Figure 18A. Time History of an In-Flight Transition; Phase II to Phase I.

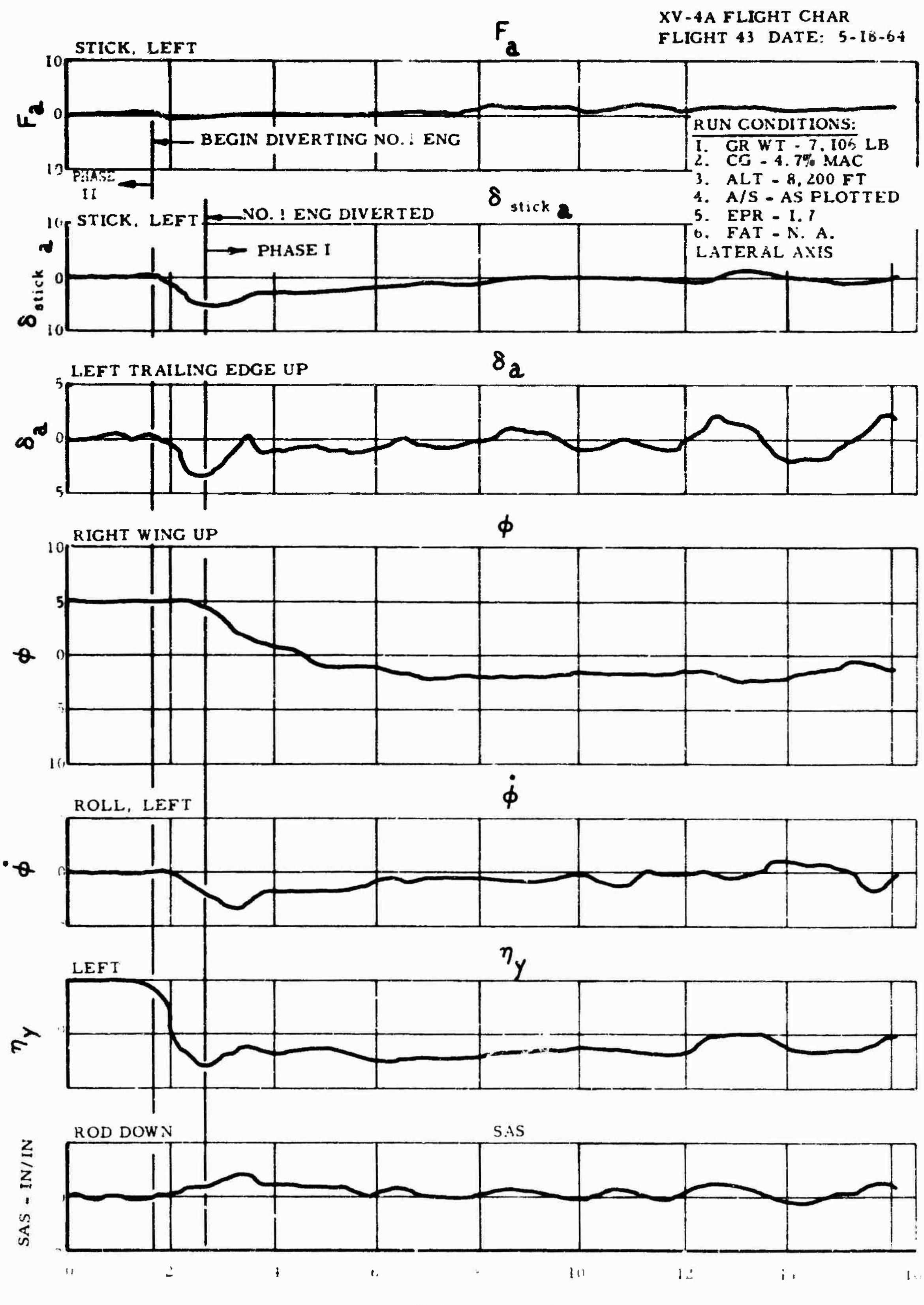


Figure 18B. Time History of an In-Flight Transition; Phase II to Phase I.

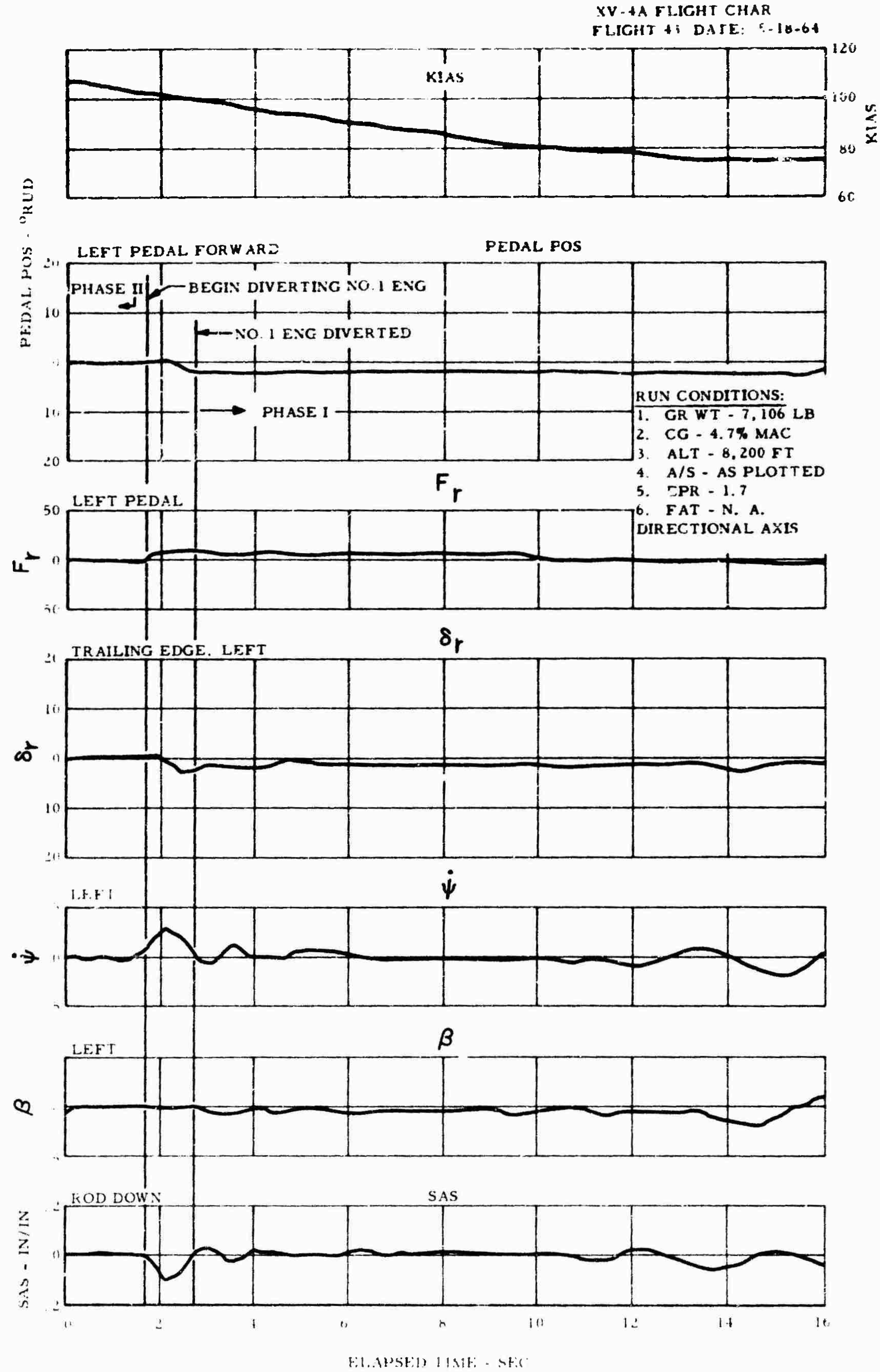


Figure 18C. Time History of an In-Flight Transition; Phase II to Phase I.

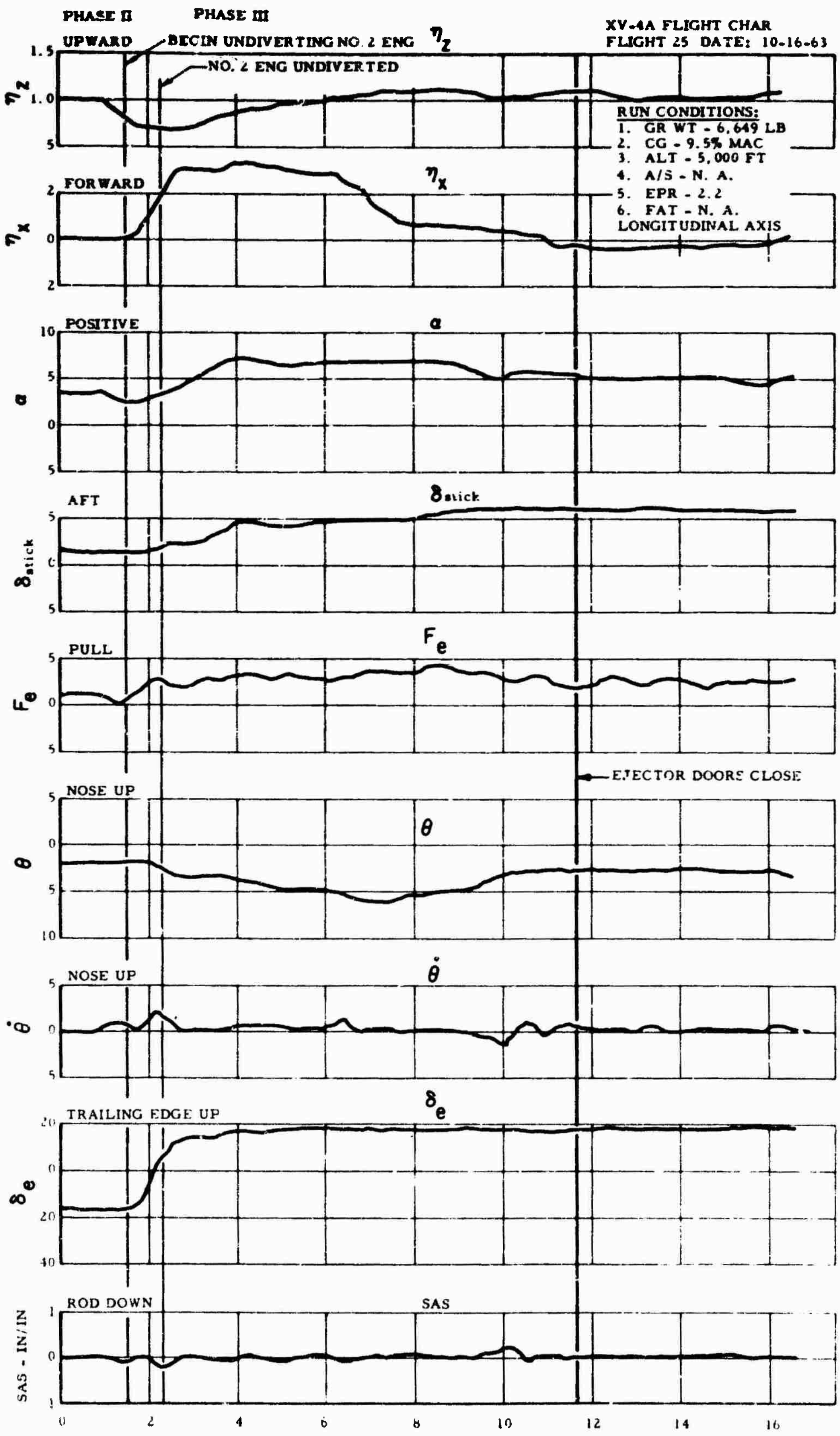


Figure 19A. Time History of an In-Flight Transition; Phase II to Phase III to Conventional Flight.

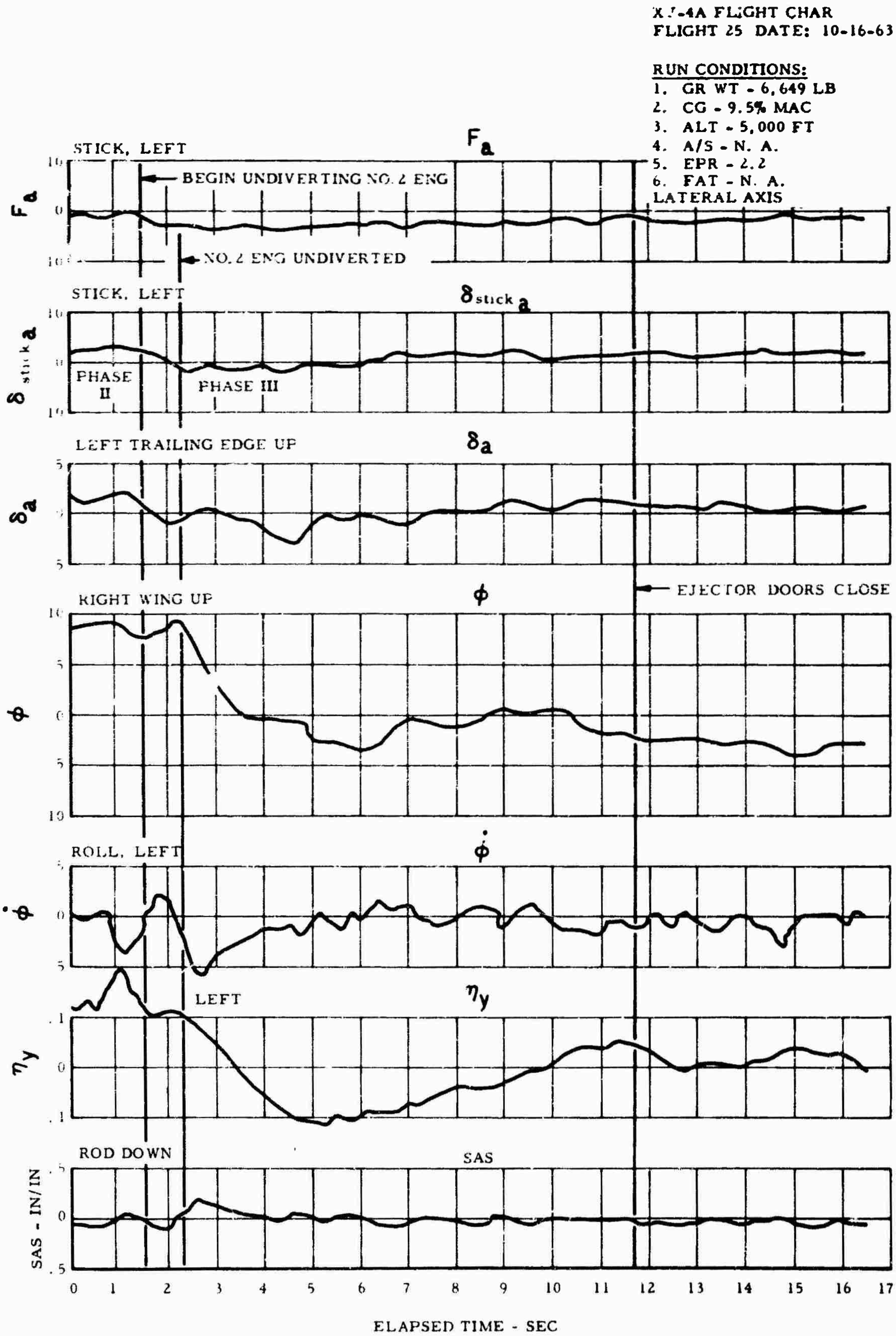


Figure 19B. Time History of an In-Flight Transition; Phase II to Phase III to Conventional Flight.

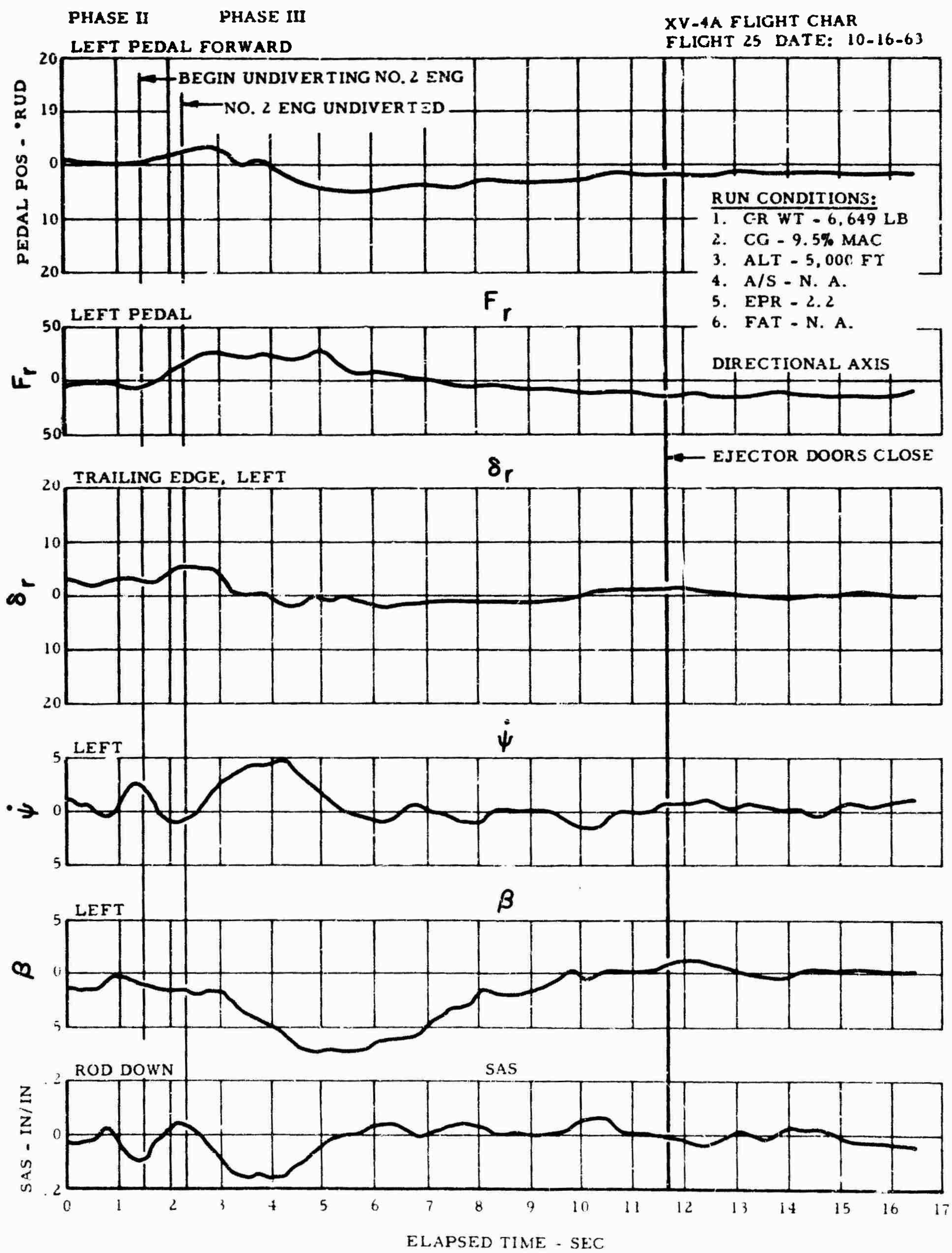


Figure 19C. Time History of an In-Flight Transition; Phase II to Phase III to Conventional Flight.

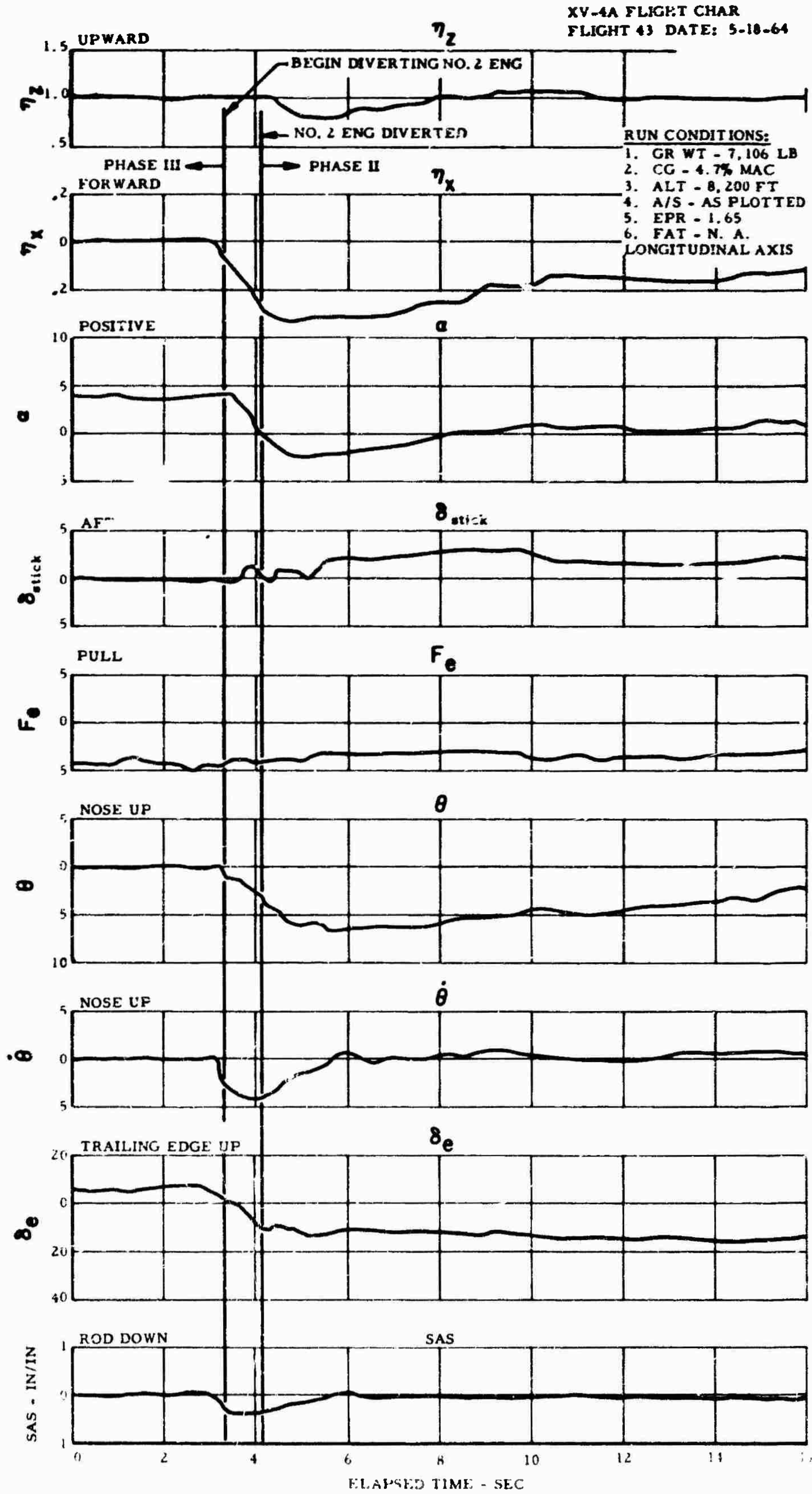


Figure 20A. Time History of an In-Flight Transition; Phase III to Phase II.

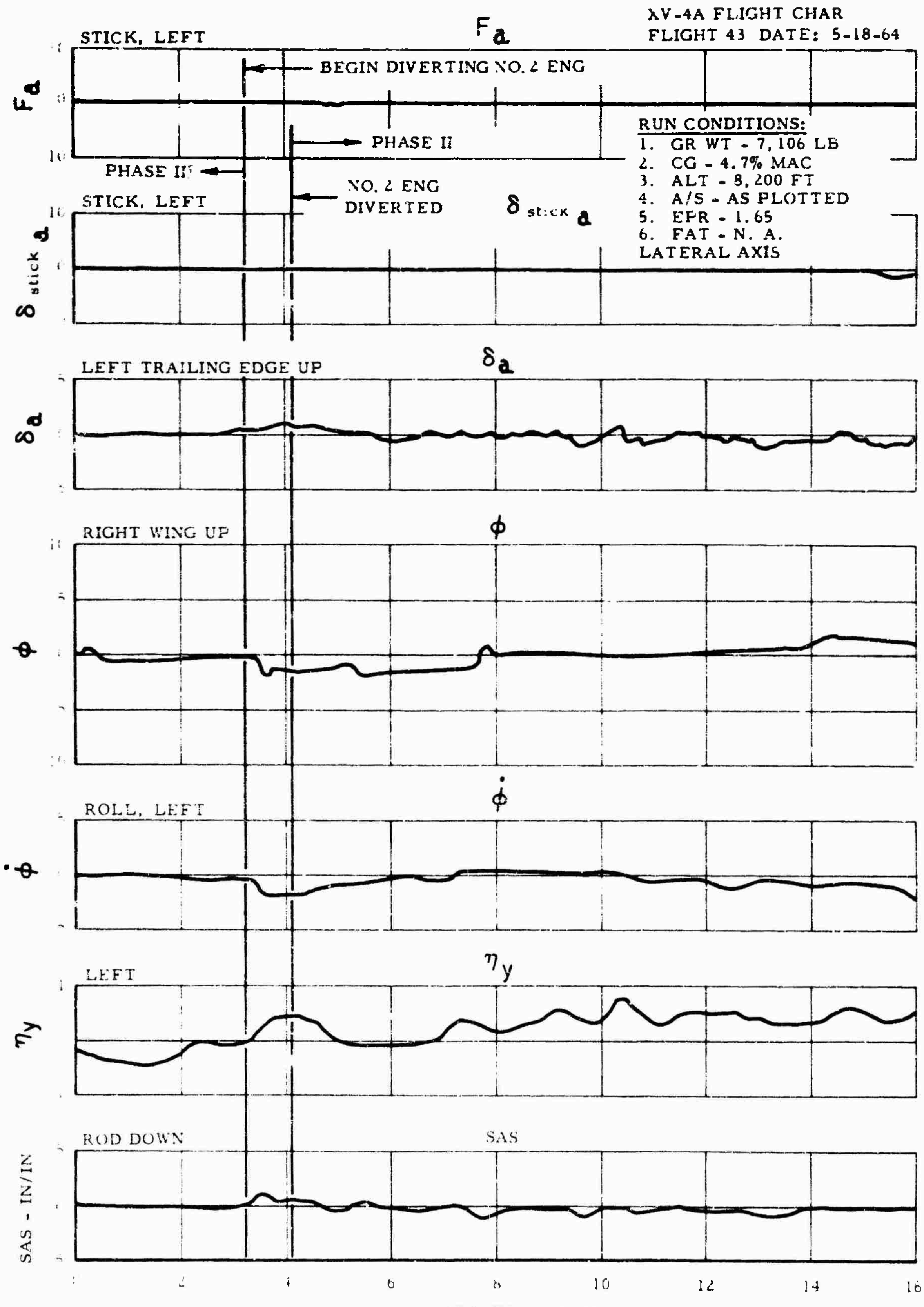


Figure 20B. Time History of an In-Flight Transition; Phase III to Phase II.

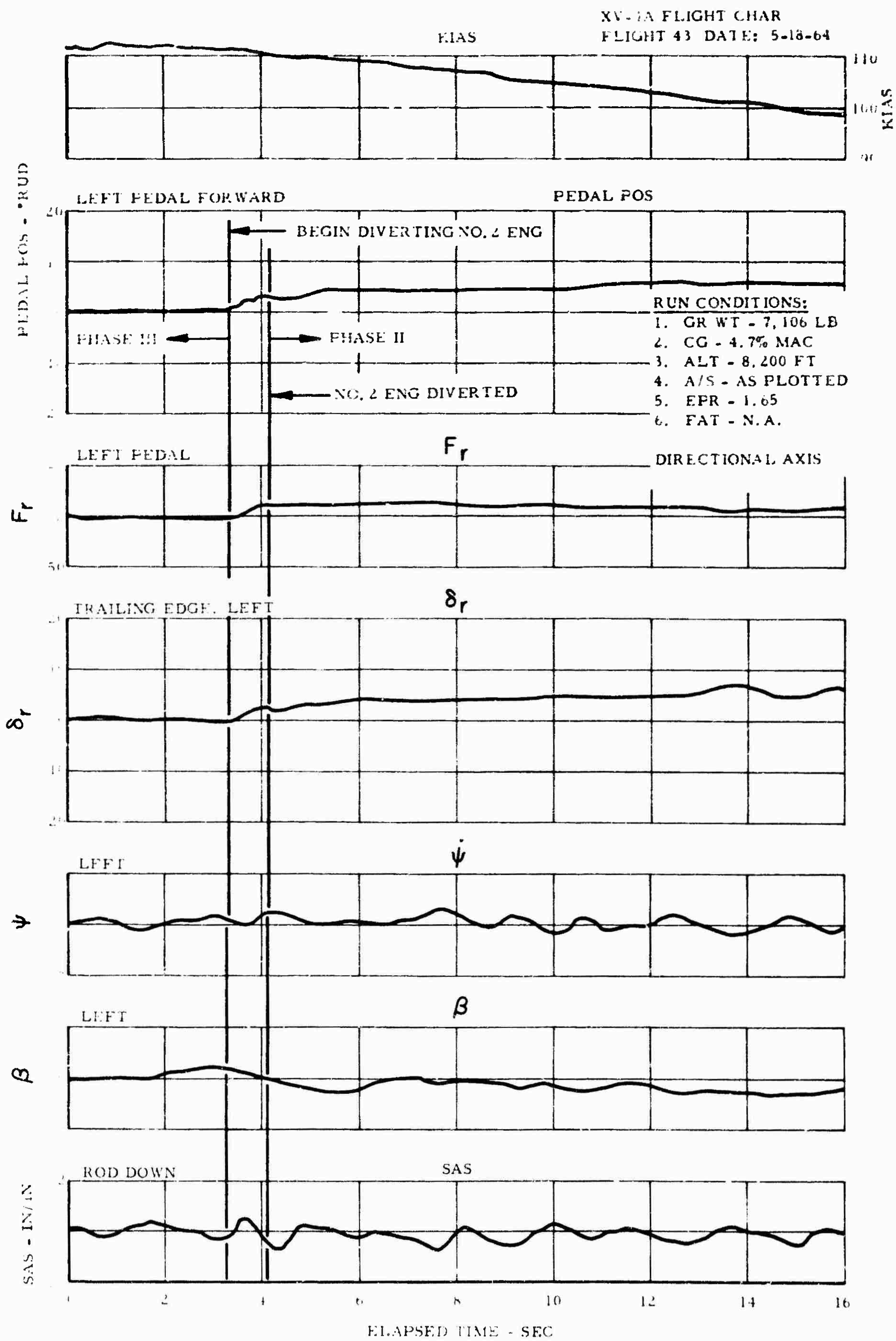


Figure 20C. Time History of an In-Flight Transition; Phase III to Phase II.

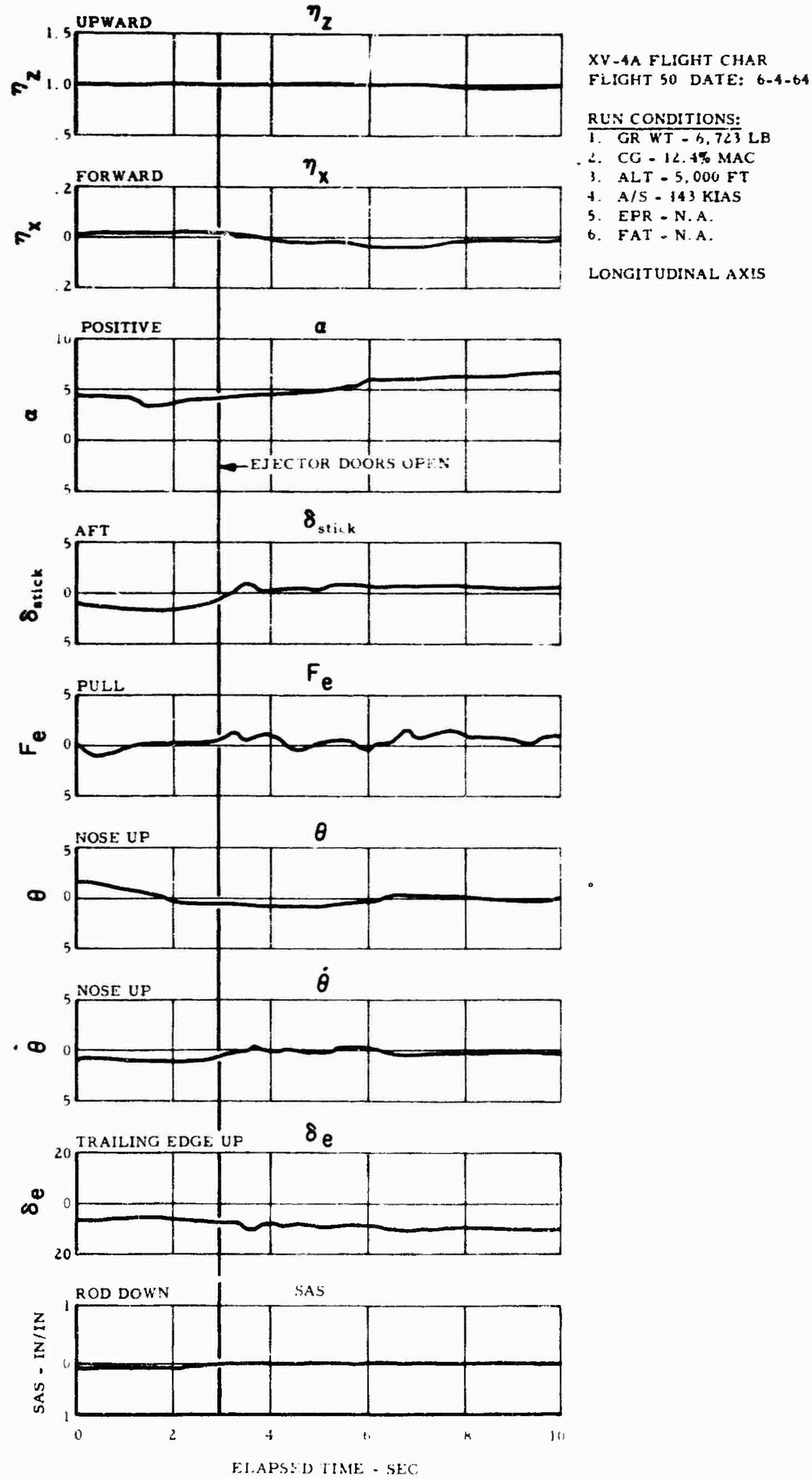
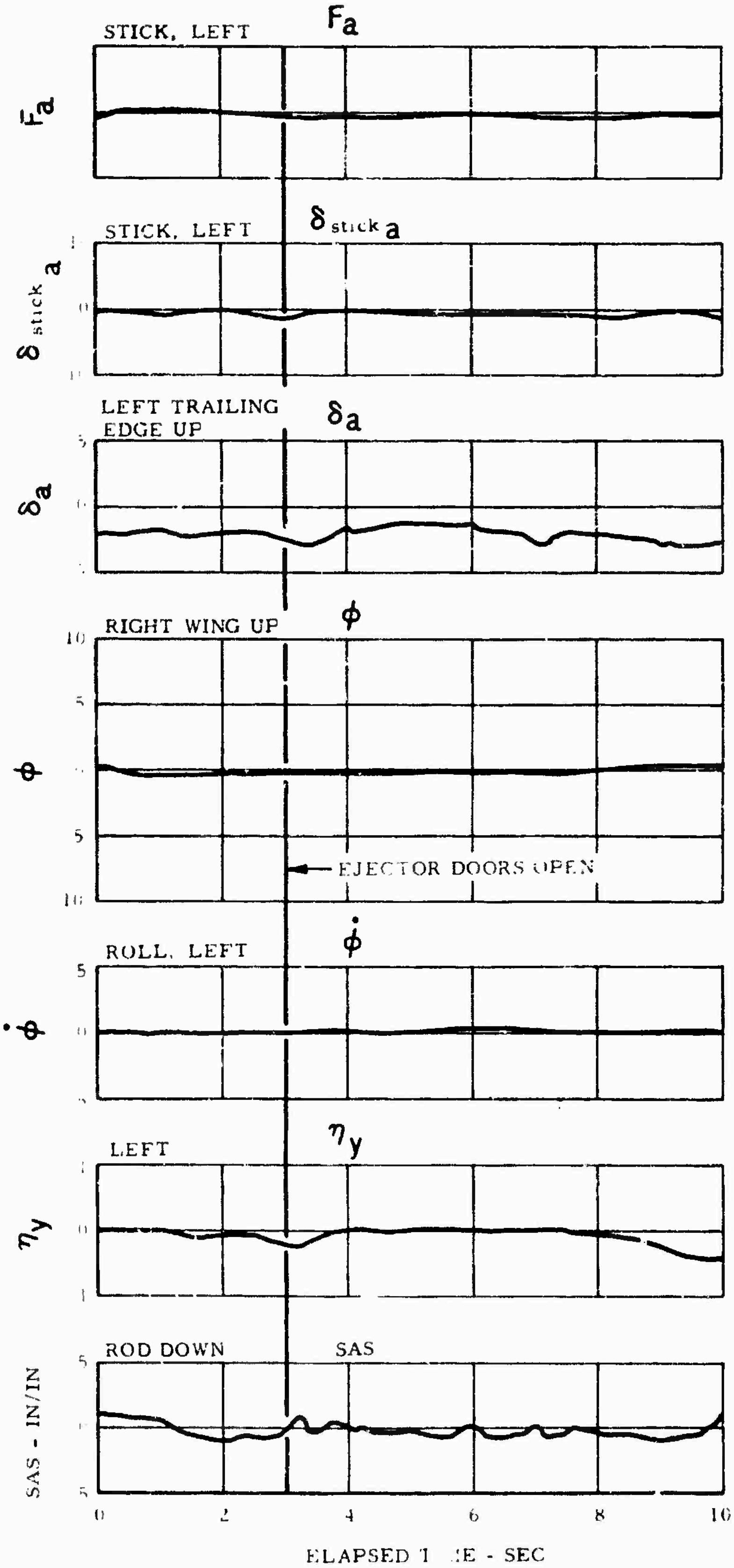


Figure 21A. Time History of Opening of Ejector Doors.



XV-4A FLIGHT CHAR
FLIGHT 50 DATE: 6-4-64

RUN CONDITIONS:
1. GR WT - 6,723 LB
2. CG - 12.4% MAC
3. ALT - 5,000 FT
4. A/S - 143 KIAS
5. EPR - N. A.
6. FAT - N. A.

LATERAL AXIS

Figure 21B. Time History of Opening of Ejector Doors.

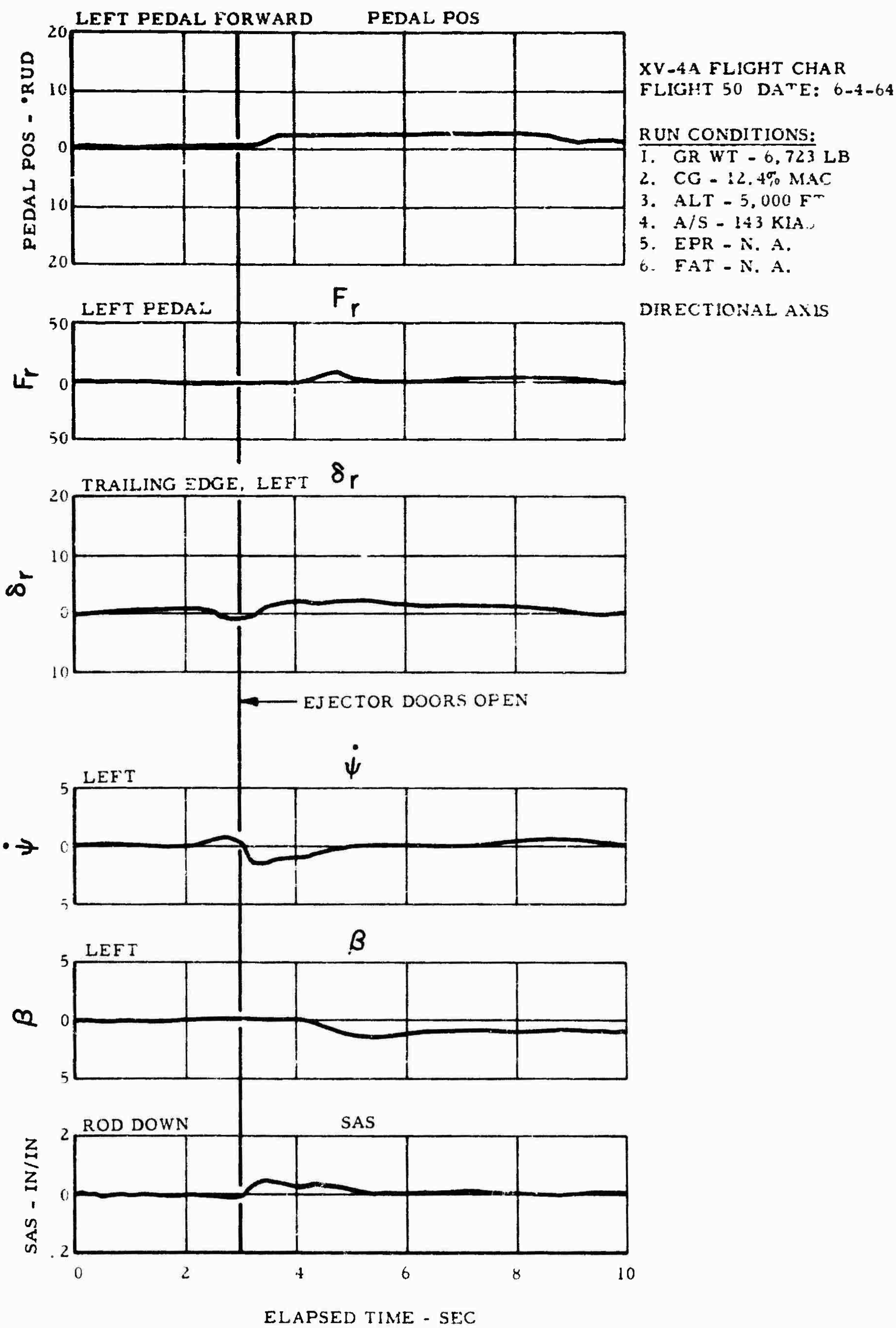


Figure 21C. Time History of Opening of Ejector Doors.

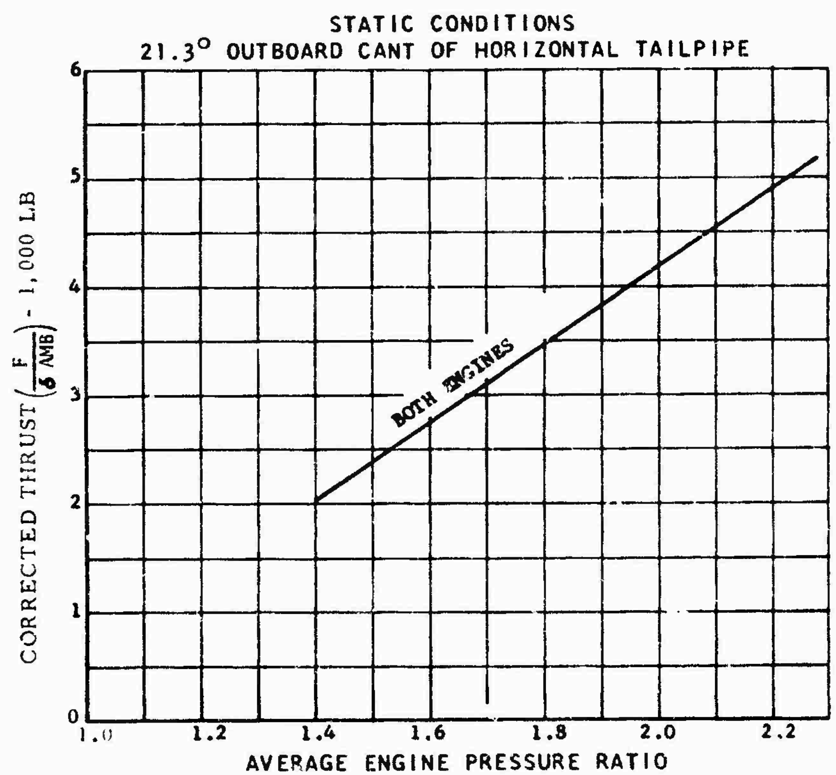


Figure 22. CTOL Propulsion System Performance.

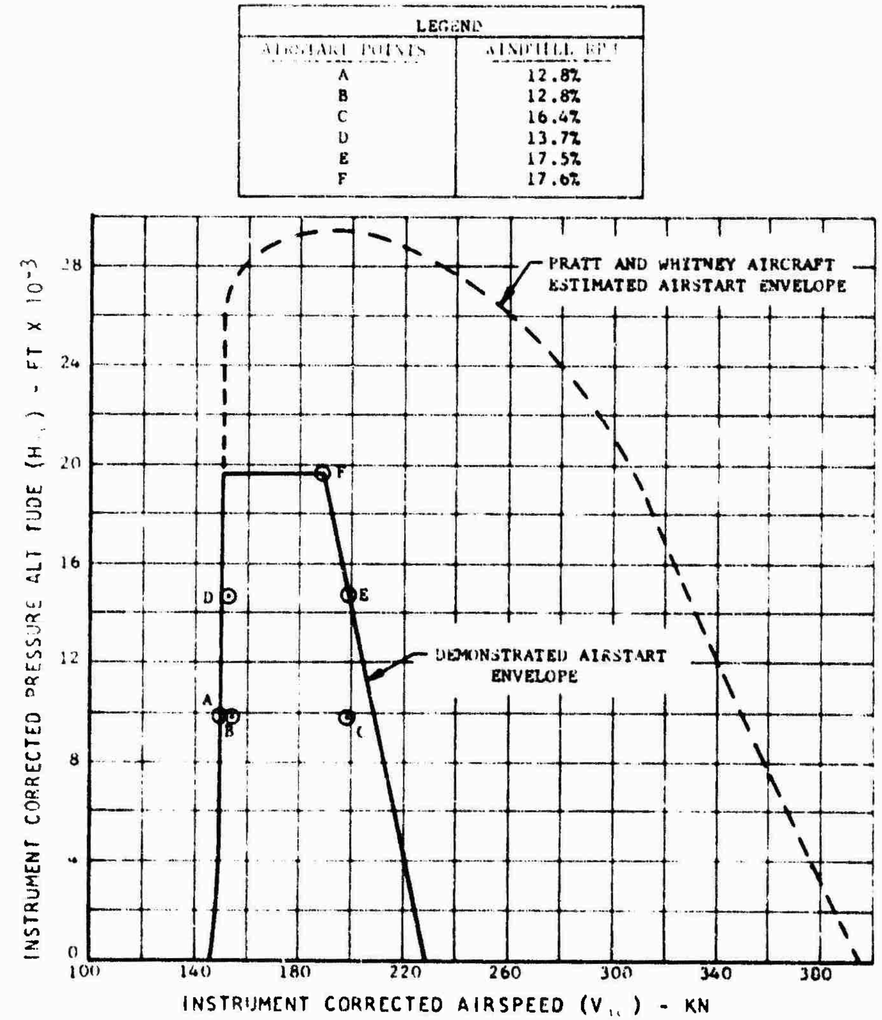


Figure 23. Airstart Envelope of JT-12A-3(LH) Engine.

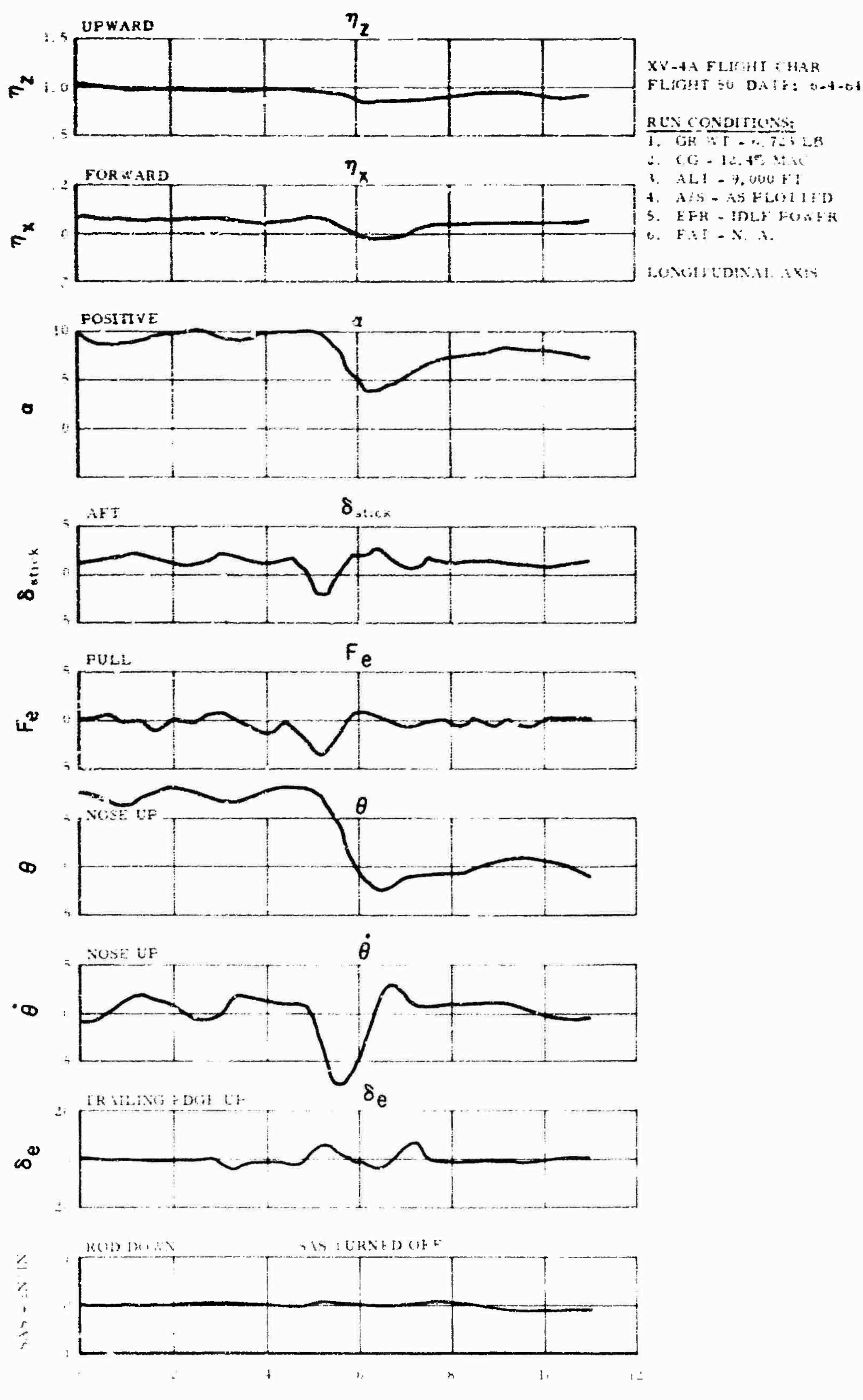
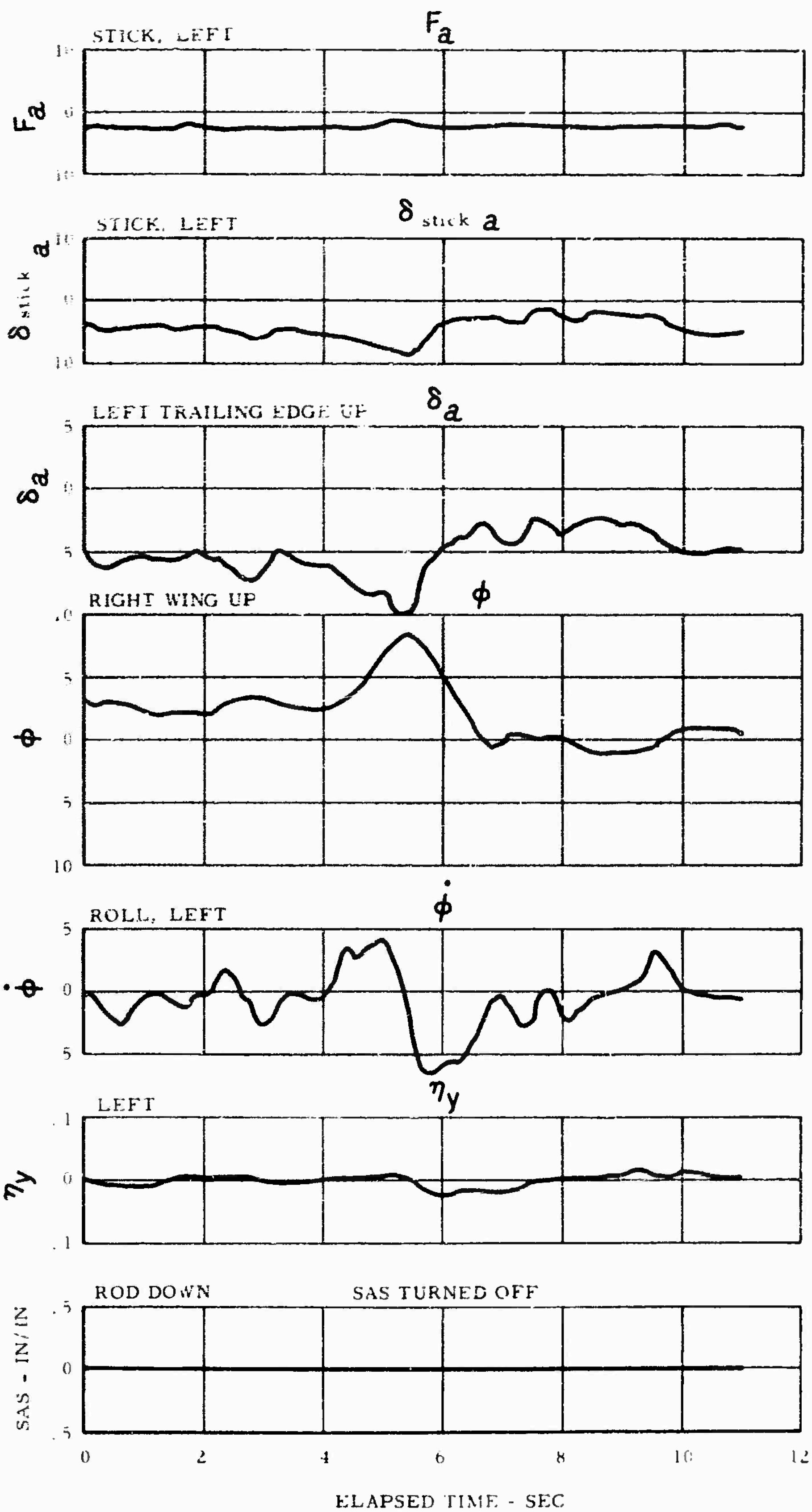


Figure 24A. Time History of a Stall in Clean Configuration.



XV-4A FLIGHT CHAR
FLIGHT 50 DATE: 6-4-64

RUN CONDITIONS:
1. GR WT - 6,723 LB
2. CG - 12.4% MAC
3. ALT - 9,000 FT
4. A/S - AS PLOTTED
5. EPR - IDLE POWER
6. FAT - N. A.

LATERAL AXIS

Figure 24B. Time History of a Stall in Clean Configuration.

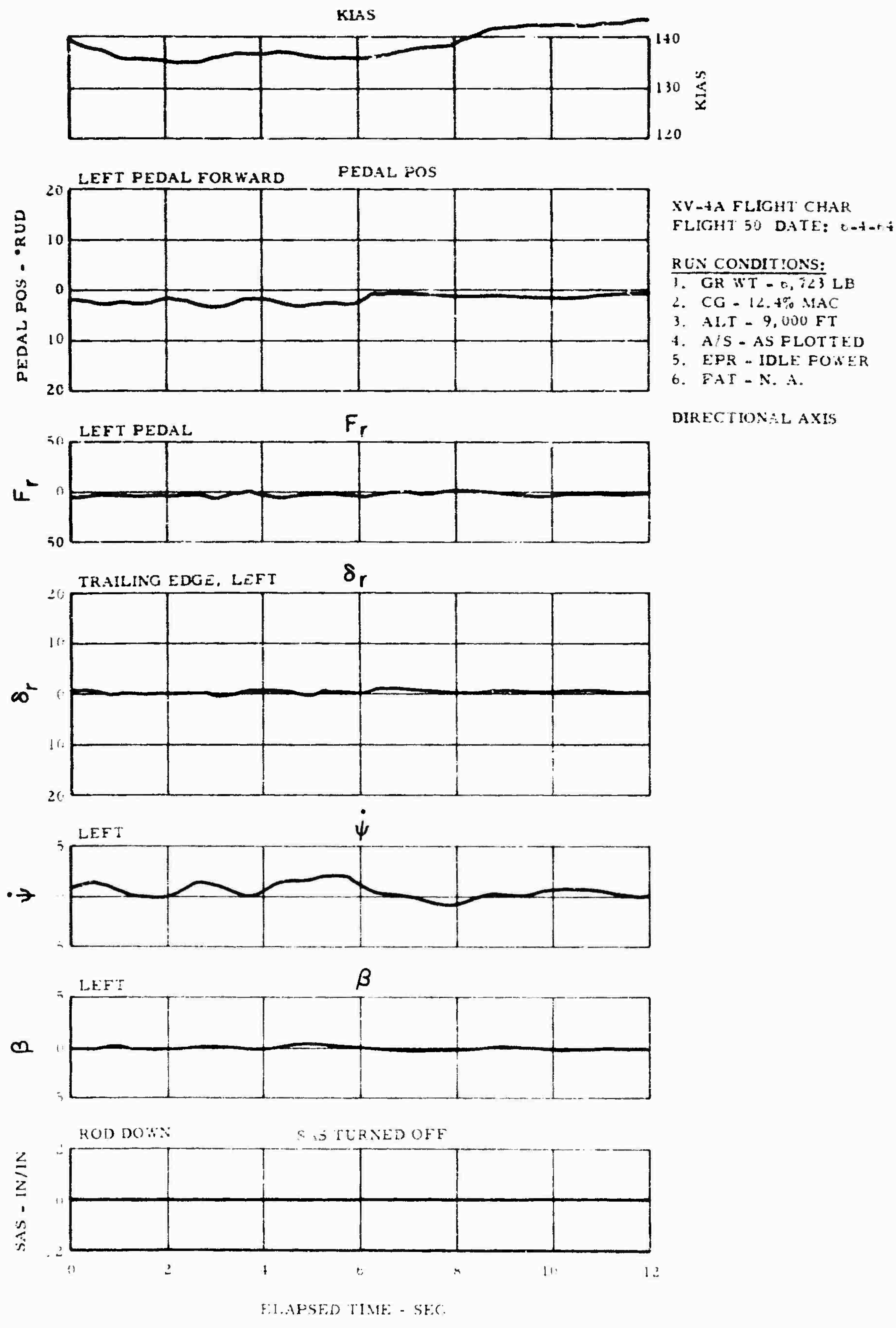


Figure 24C. Time History of a Stall in Clean Configuration.

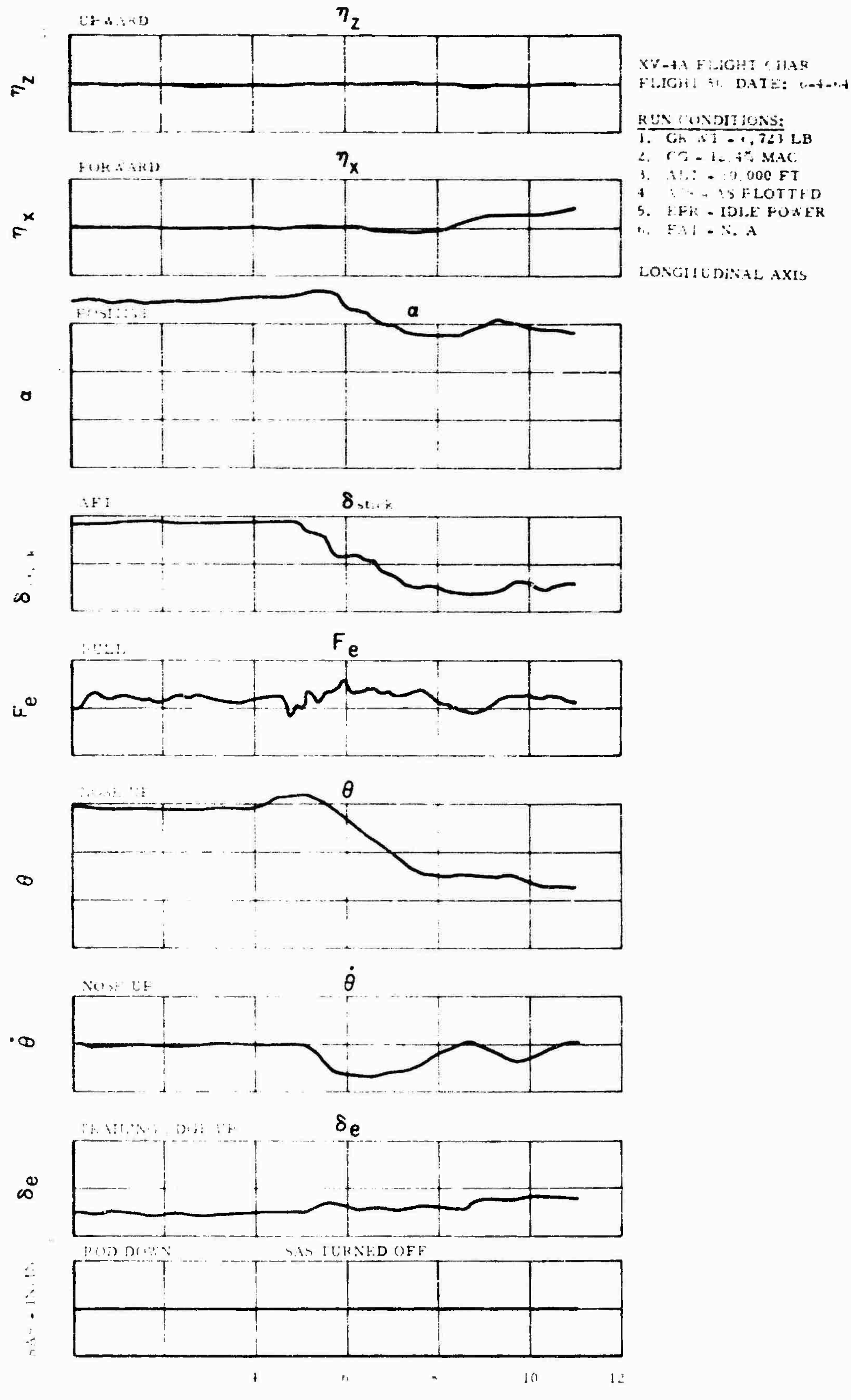


Figure 25A. Time History of a Stall in Landing Configuration.

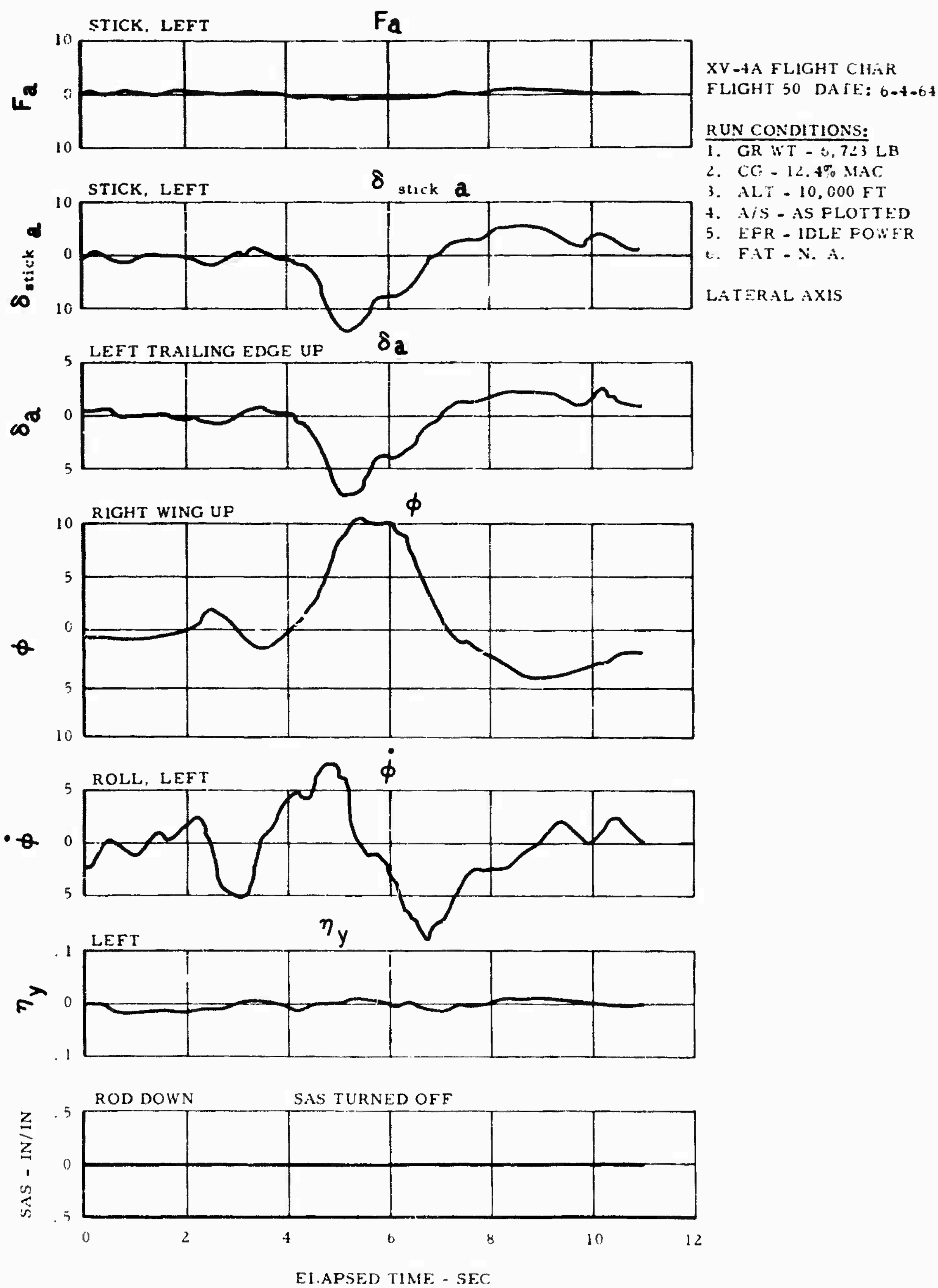


Figure 25B. Time History of a Stall in Landing Configuration.

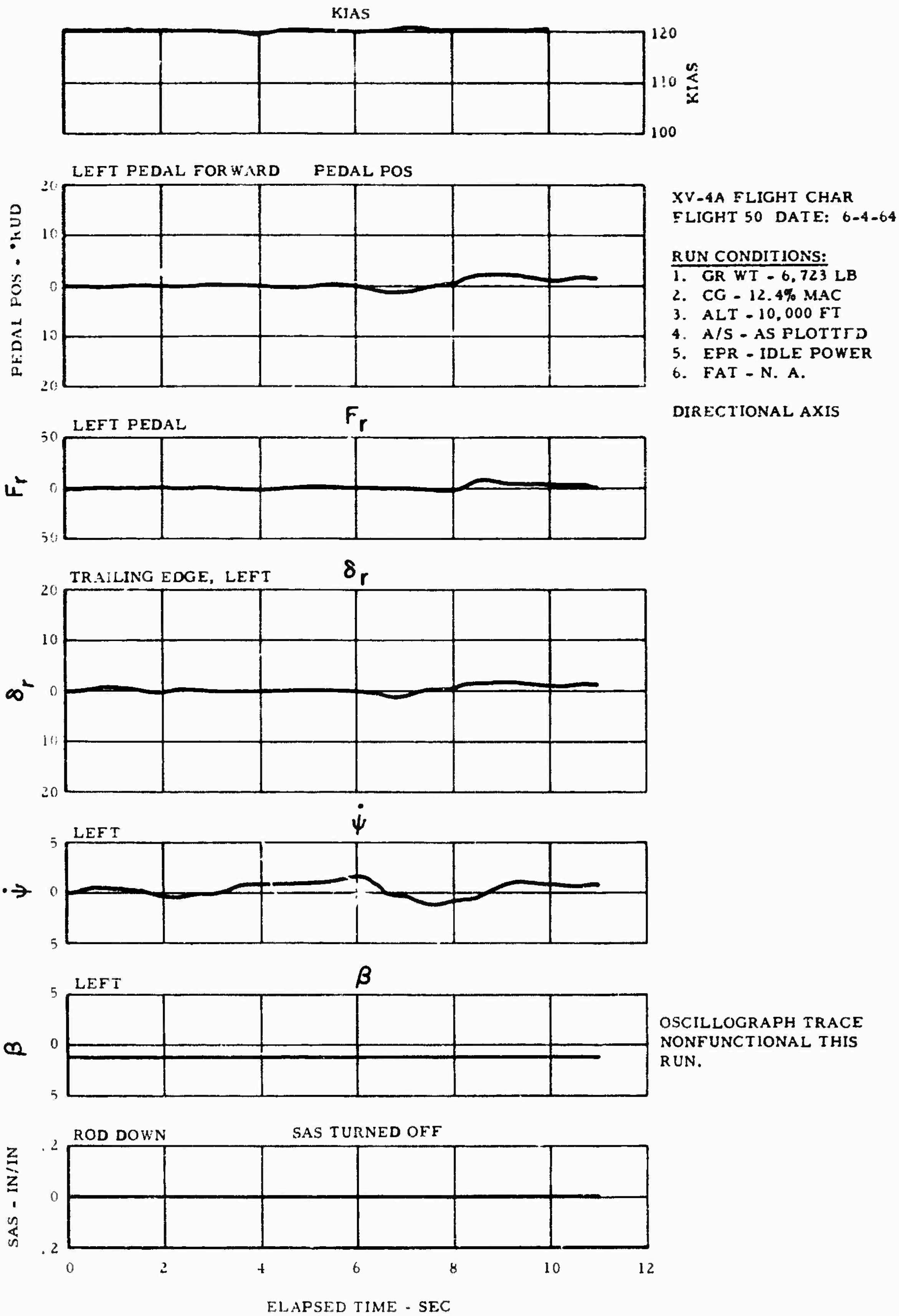


Figure 25C. Time History of a Stall in Landing Configuration.

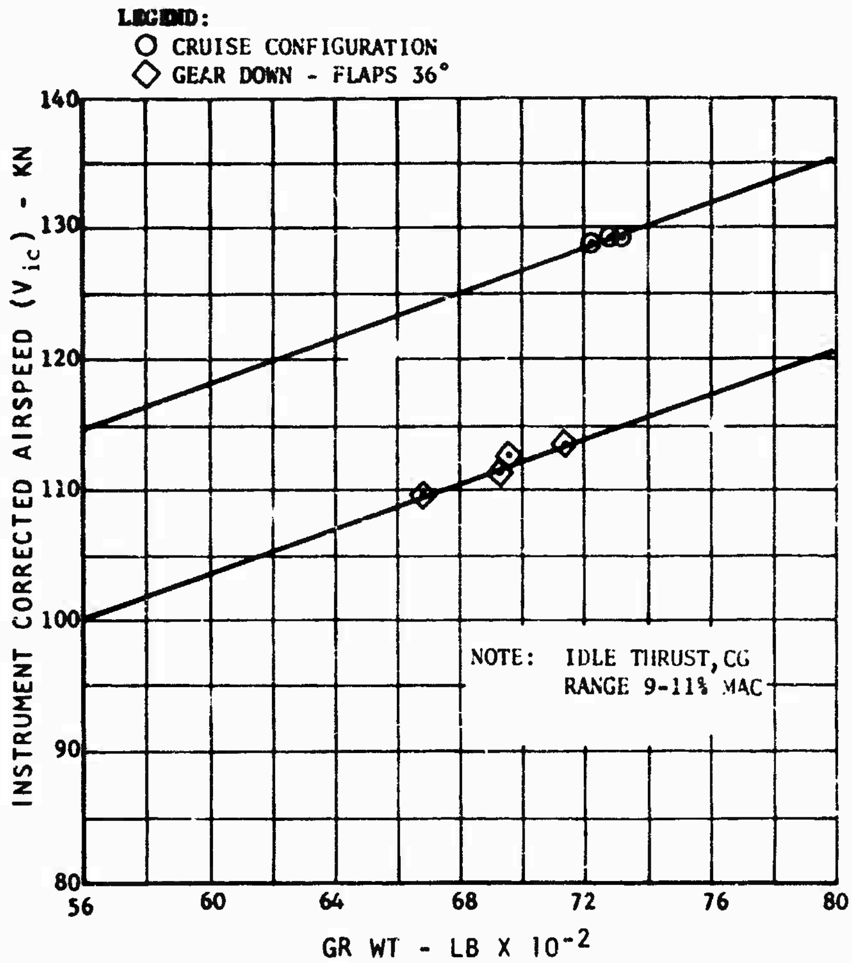


Figure 26. Demonstrated Stall Speeds.

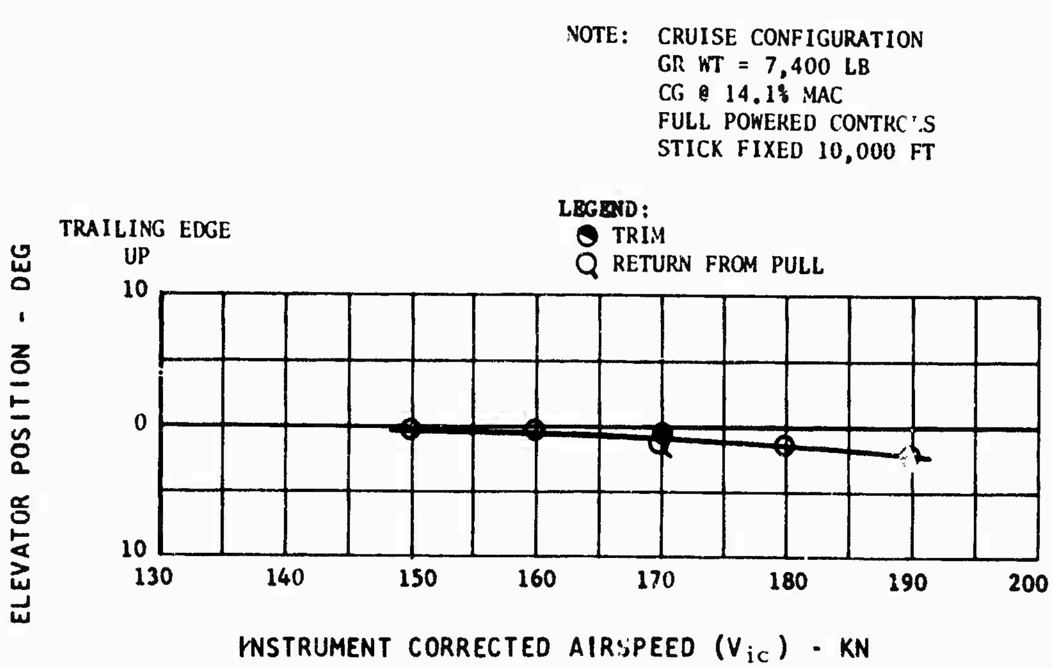


Figure 27. Static Longitudinal Stability.

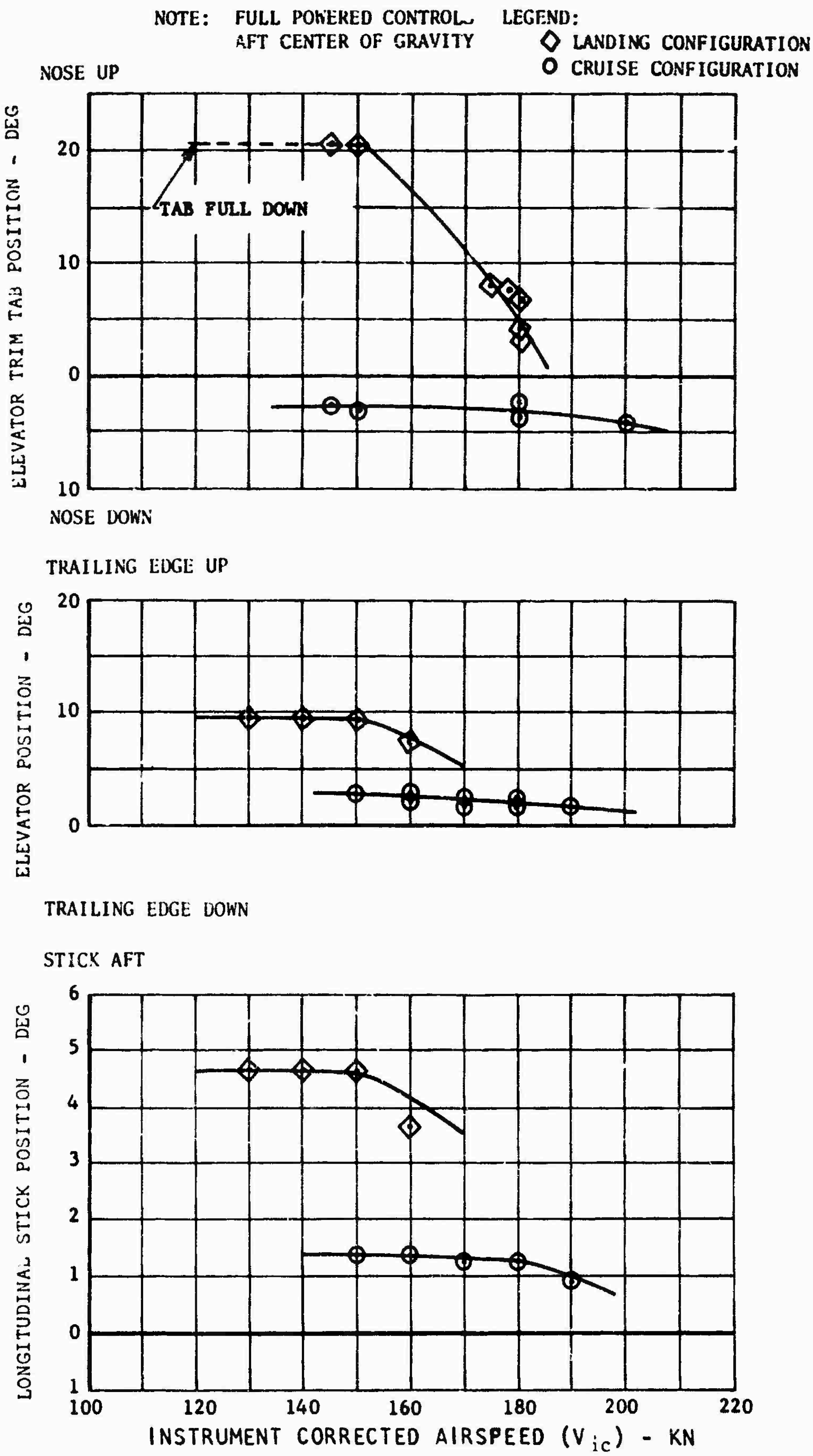


Figure 28. Longitudinal Control Effectiveness.

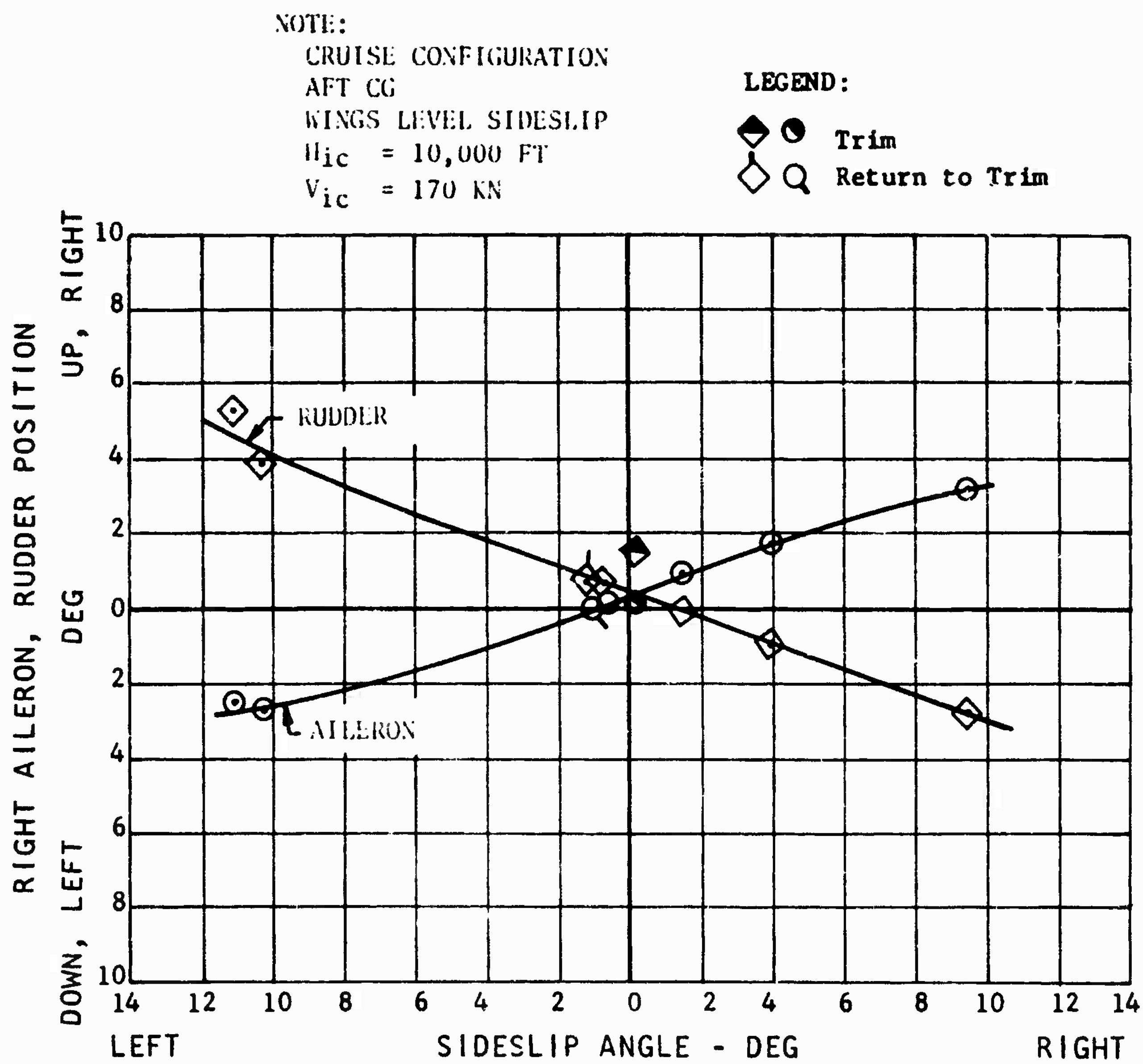


Figure 29. Lateral-Directional Stability; Cruise Configuration.

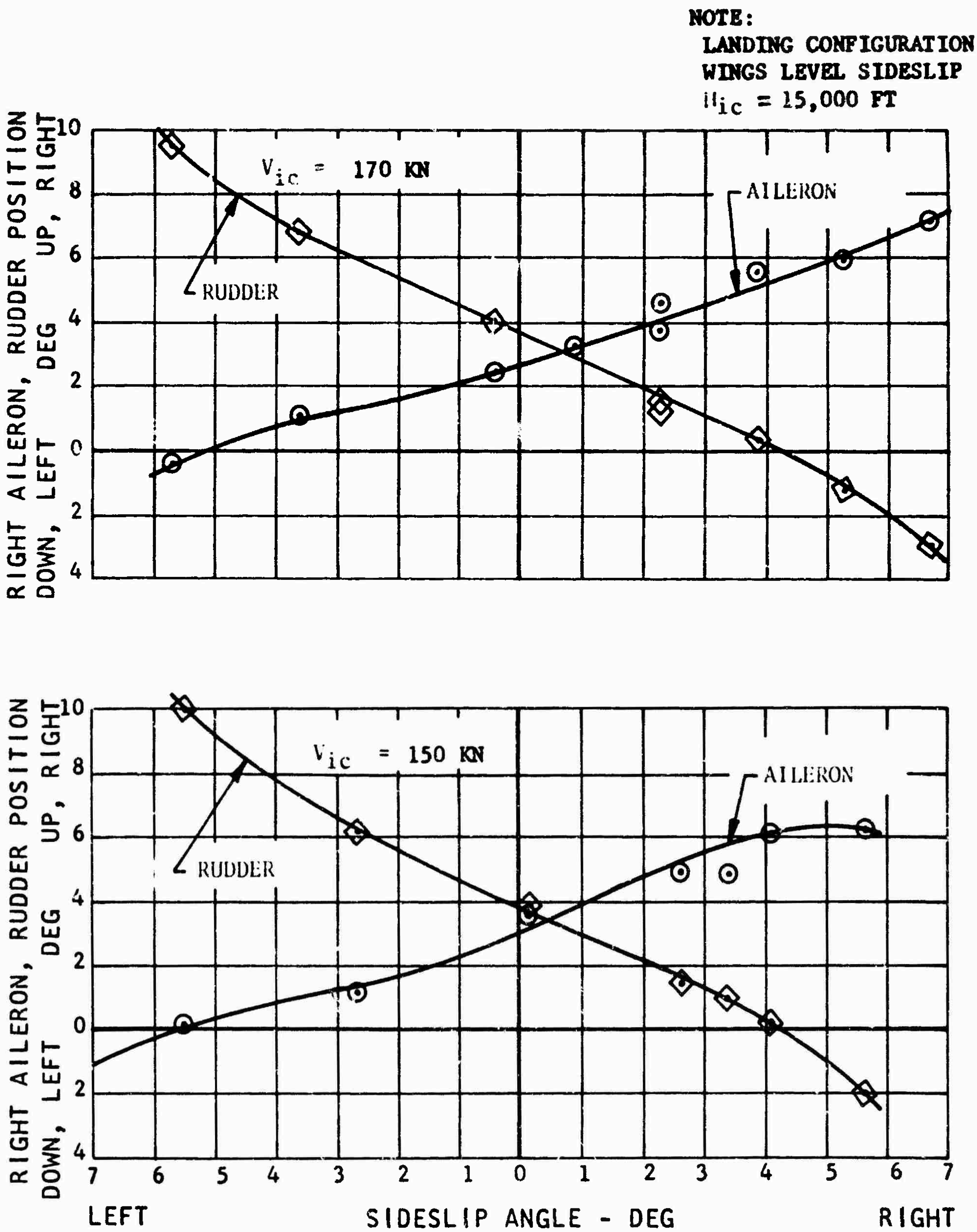


Figure 30. Lateral-Directional Stability; Landing Configuration.

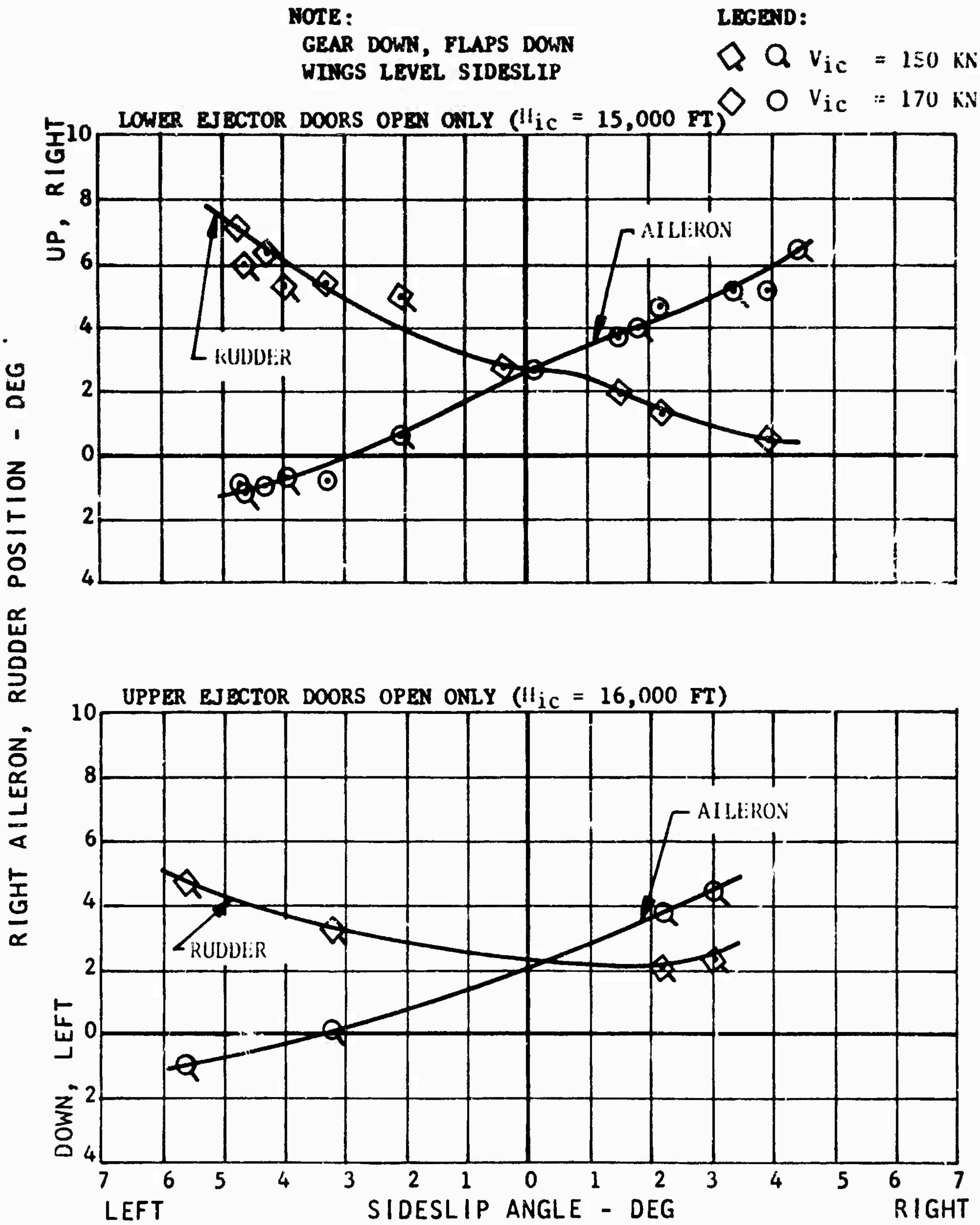


Figure 31. Lateral-Directional Stability; Ejector Doors Open.

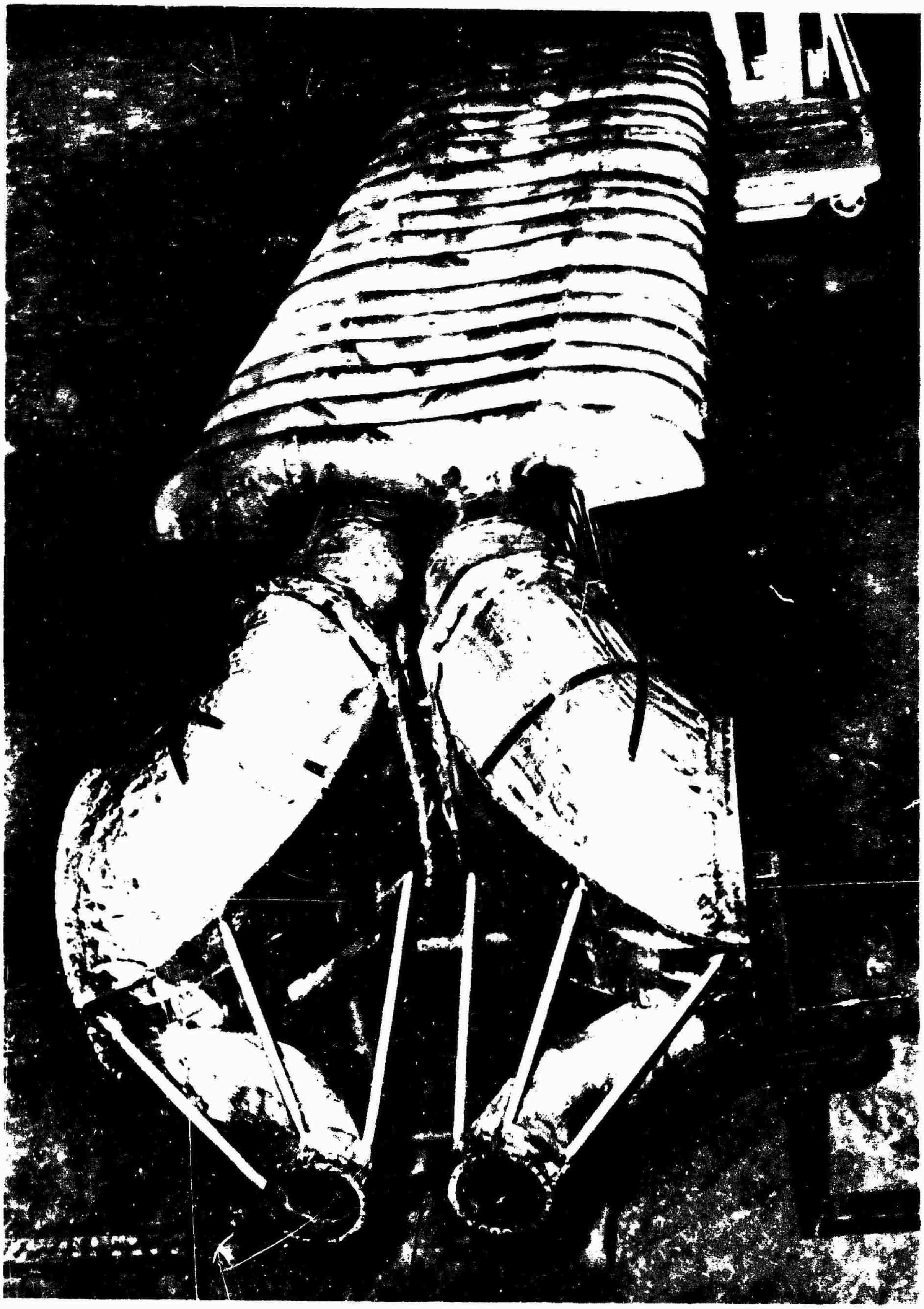


Figure 32. D Manifold Used in Tests for Ejector Configurations I and II.

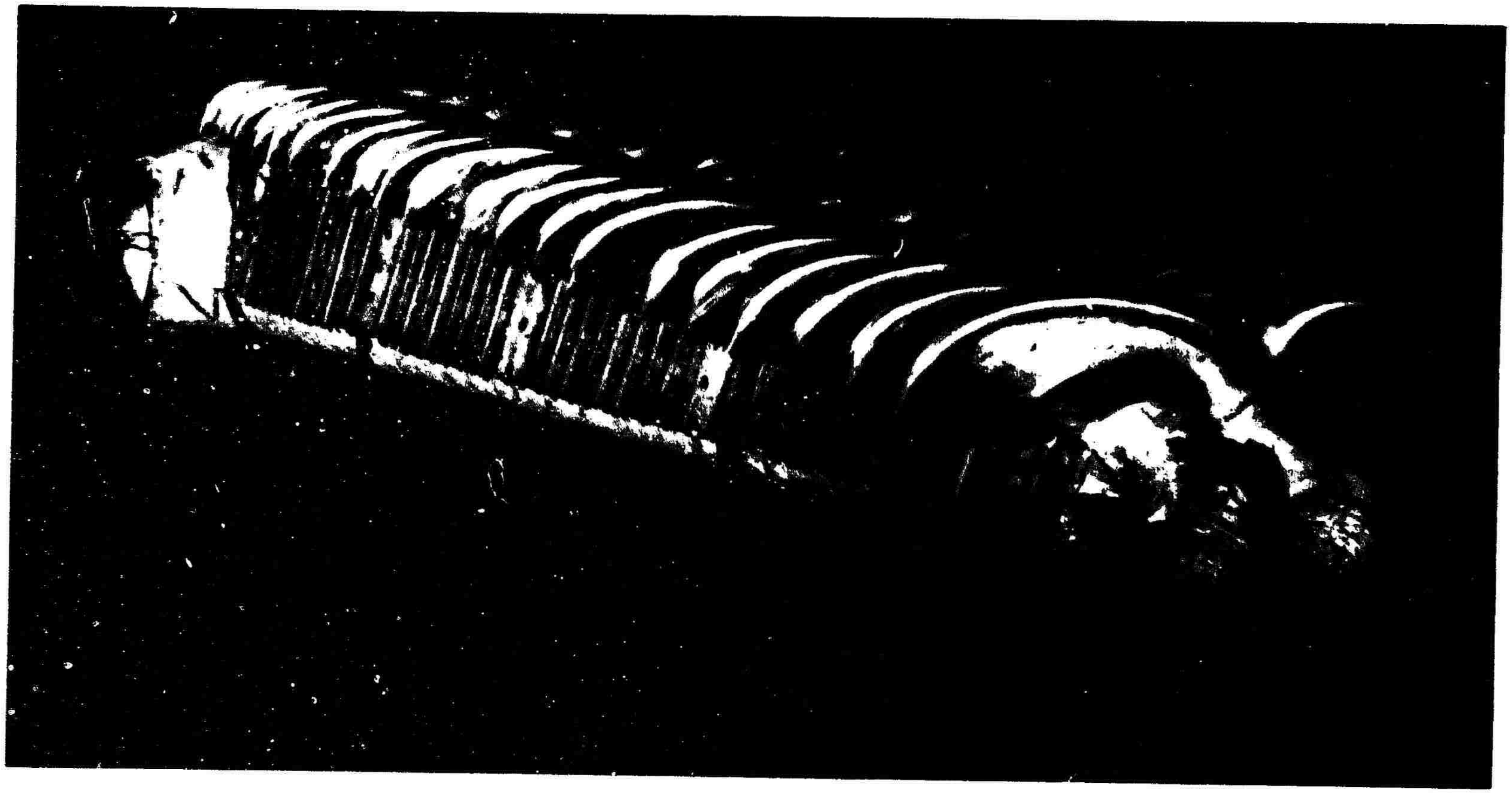


Figure 33. End View of F Manifold Used in Tests for Ejector Configurations III, IV, and V.

Figure 34. Top View of F Manifold.

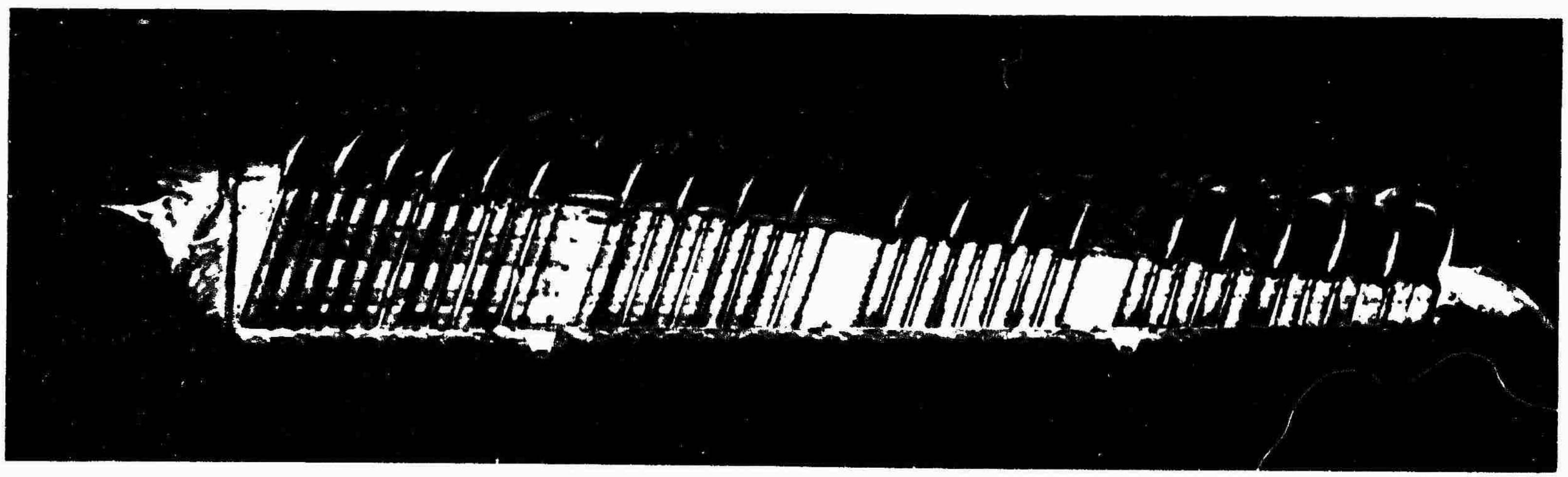


Figure 35. Side View of F Manifold.

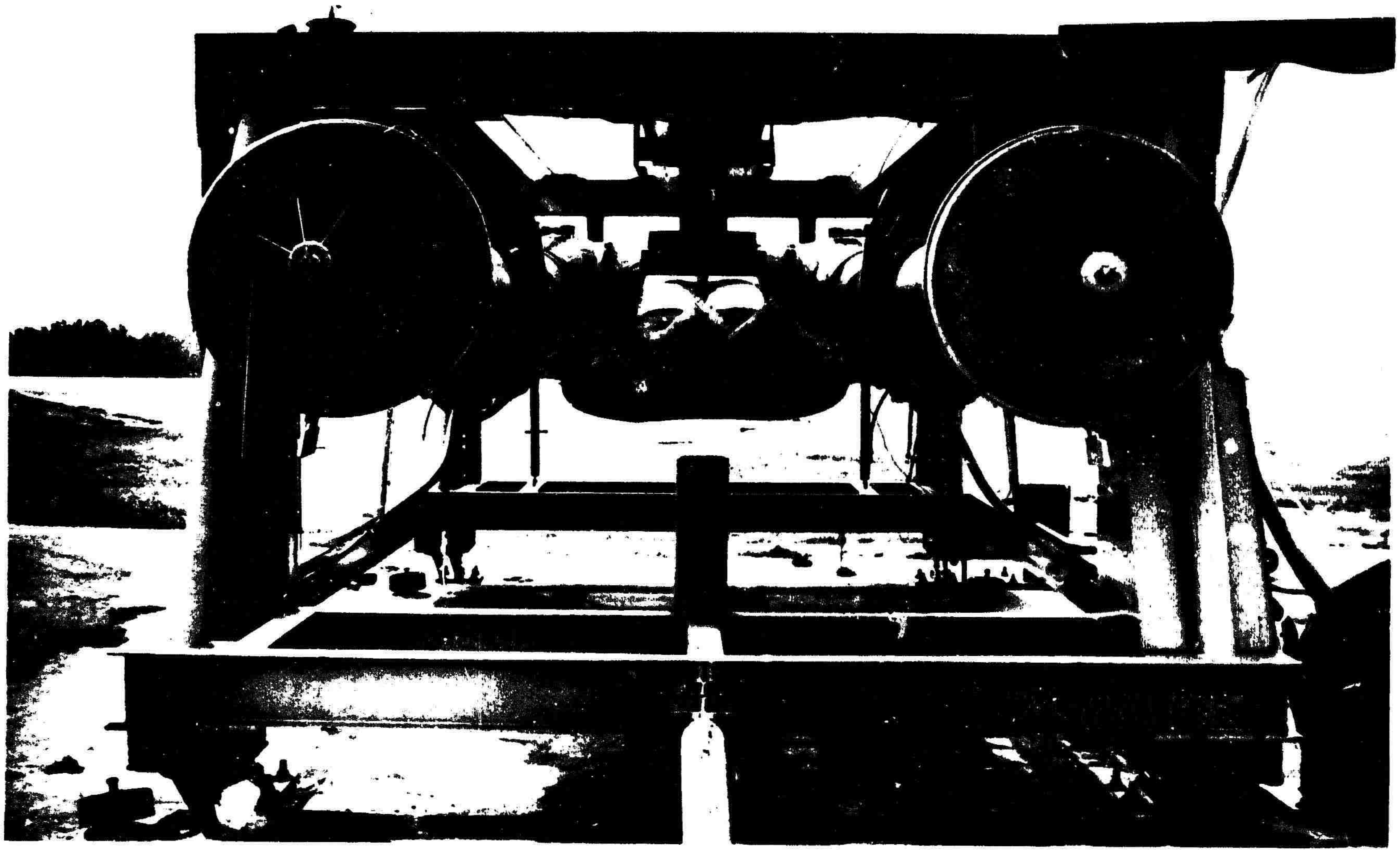


Figure 36. End View of Test Rig as Used To Obtain Manifold Nozzle Thrust.

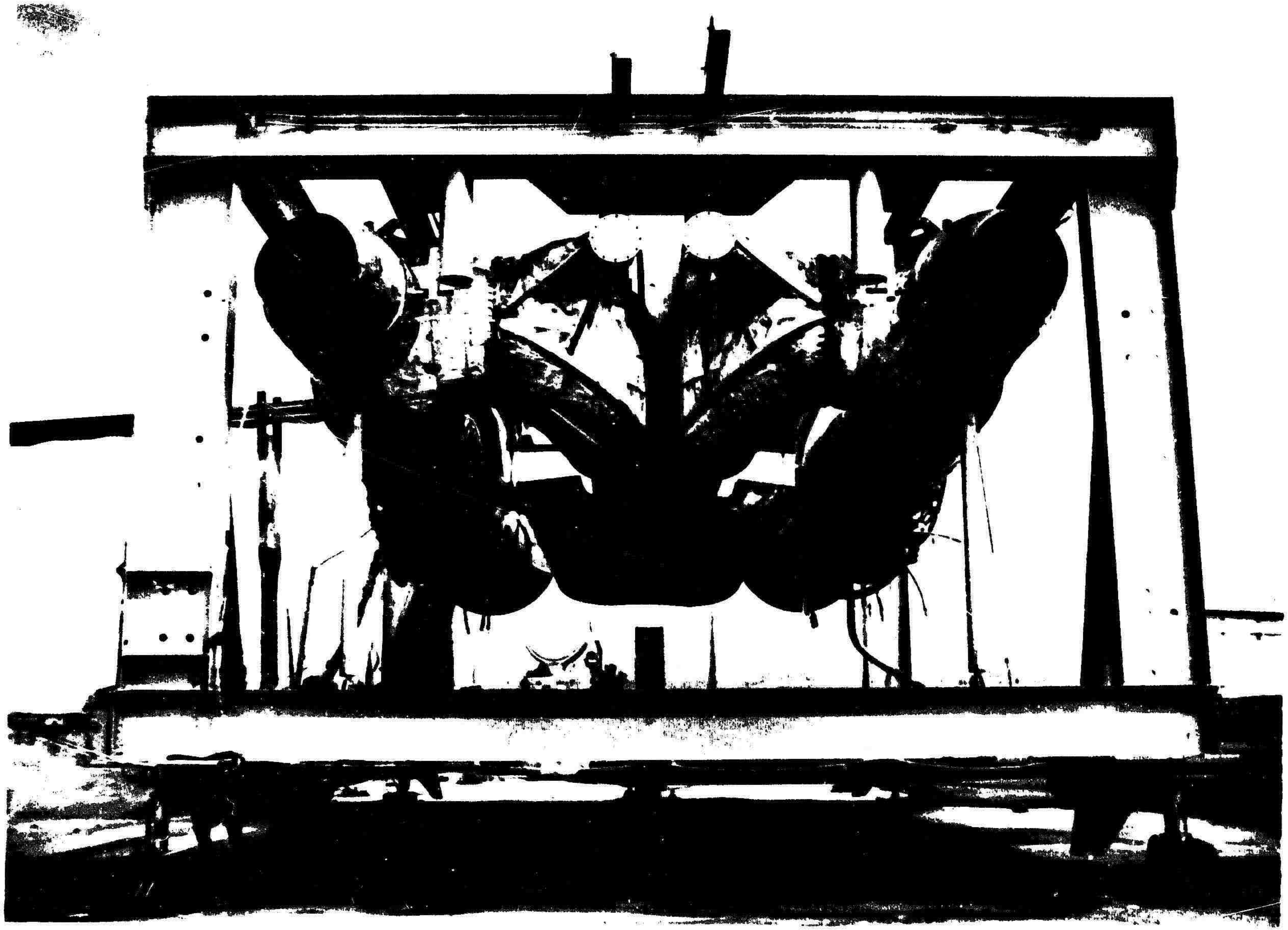


Figure 37. Rear View of Test Rig as Used To Obtain Manifold Nozzle Thrust.

Figure 38. Inlet for Ejector Configurations I, II, and III.

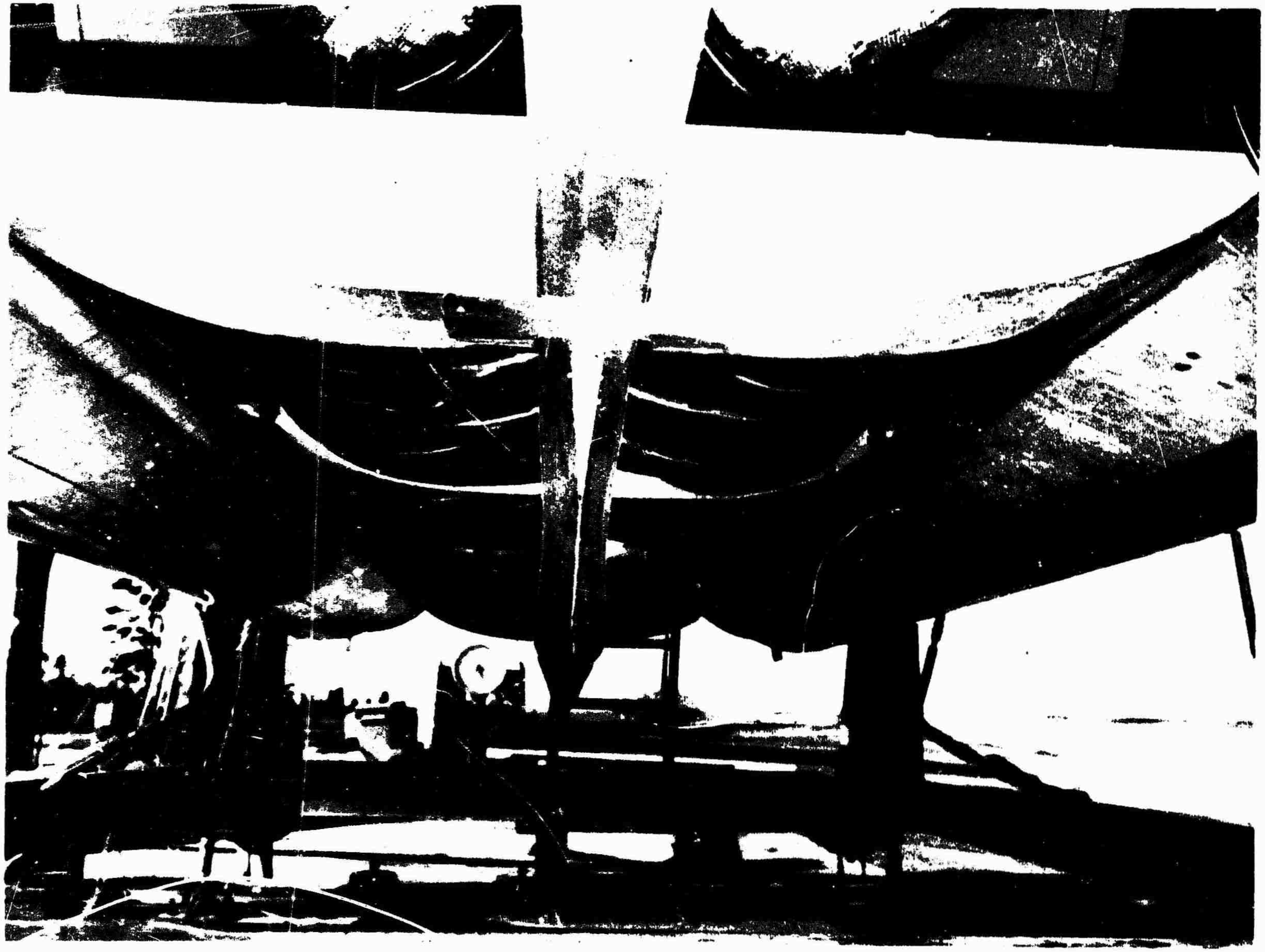


Figure 39. Inlet for Ejector Configuration V.

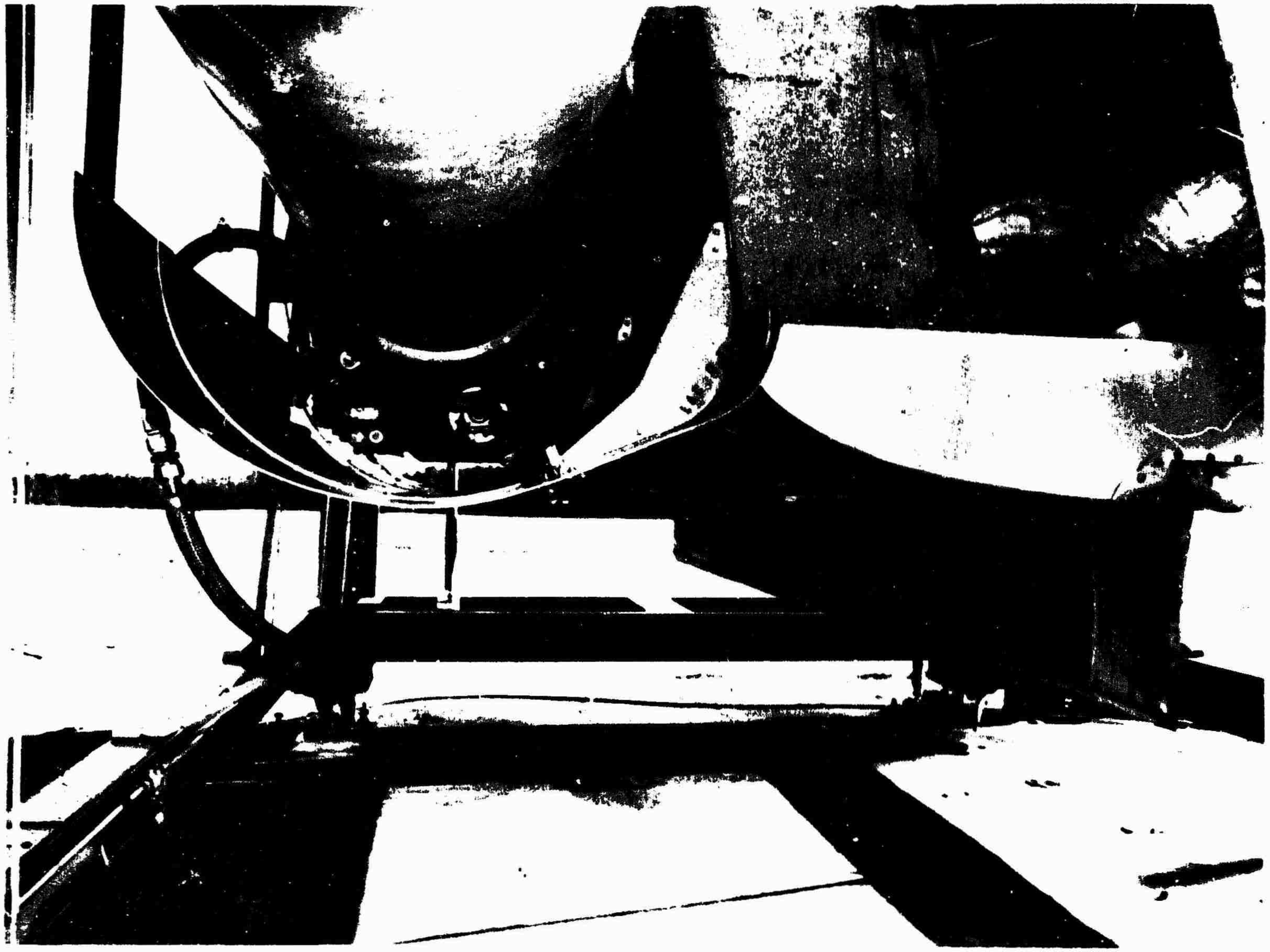


Figure 40. End View of Inlet for Ejector Configuration V.

Figure 41. Side View of Inlet for Ejector Configuration V.

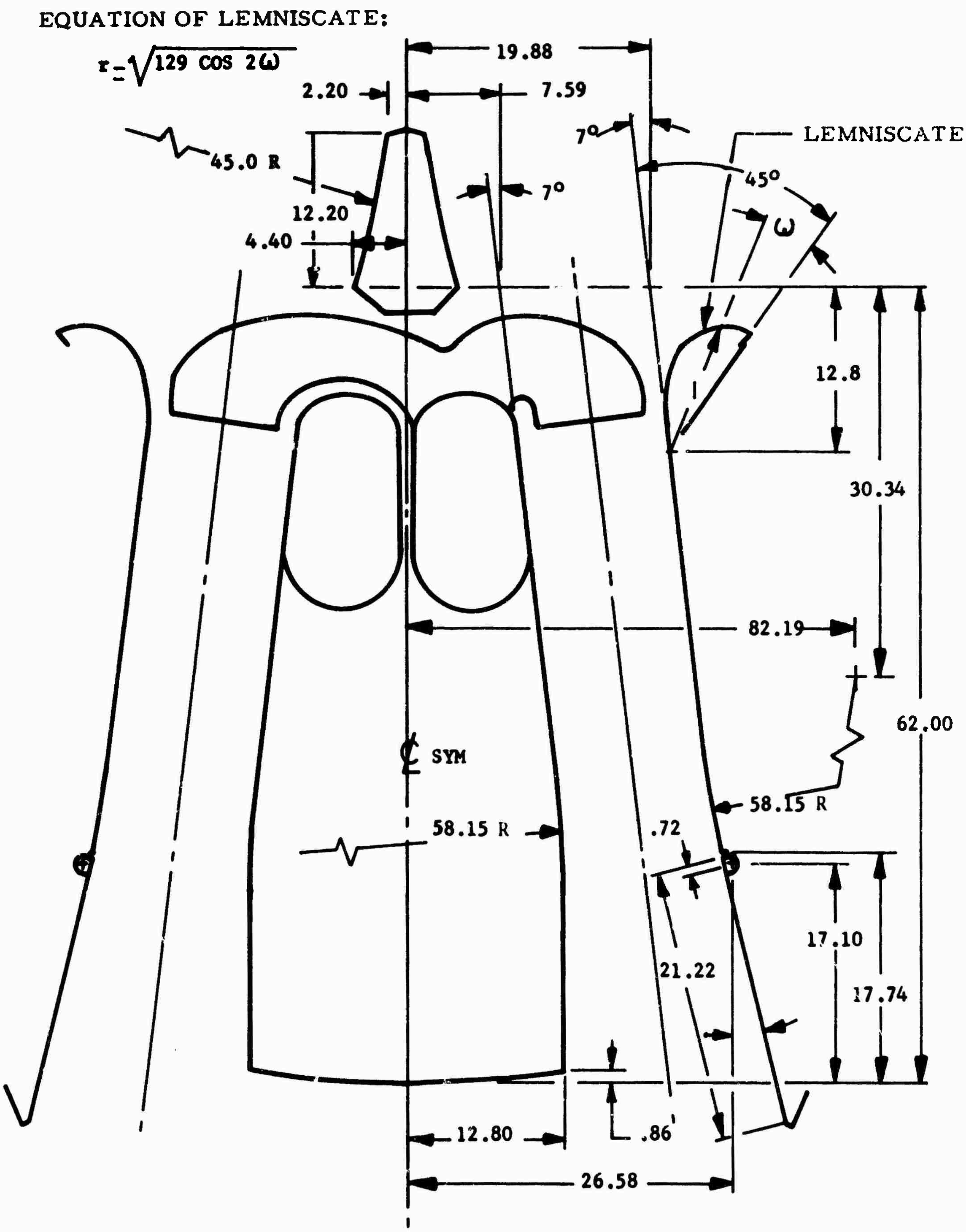


Figure 42. Basic Ejector Cross Section With Dimensions.

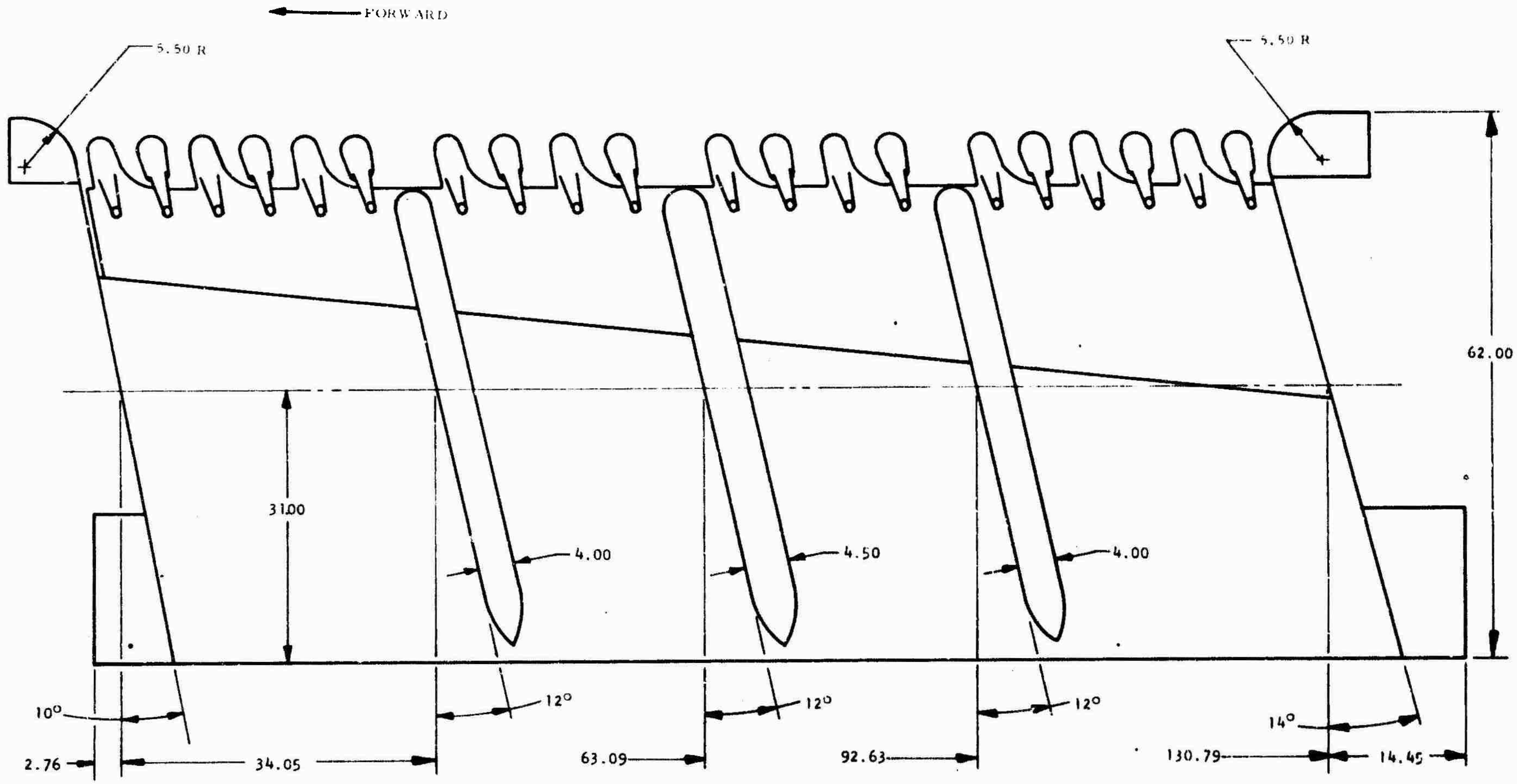


Figure 43. Basic Ejector Side View.

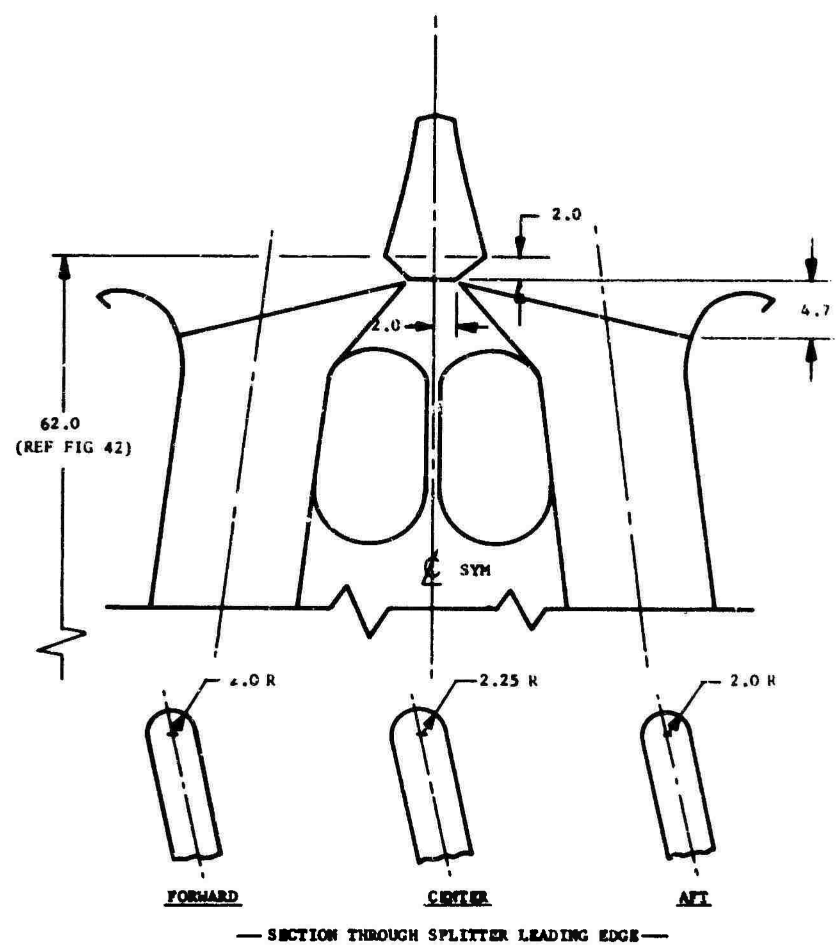


Figure 44. Ejector Splitter Leading-Edge Cross Sections for Configurations I, II, and III.

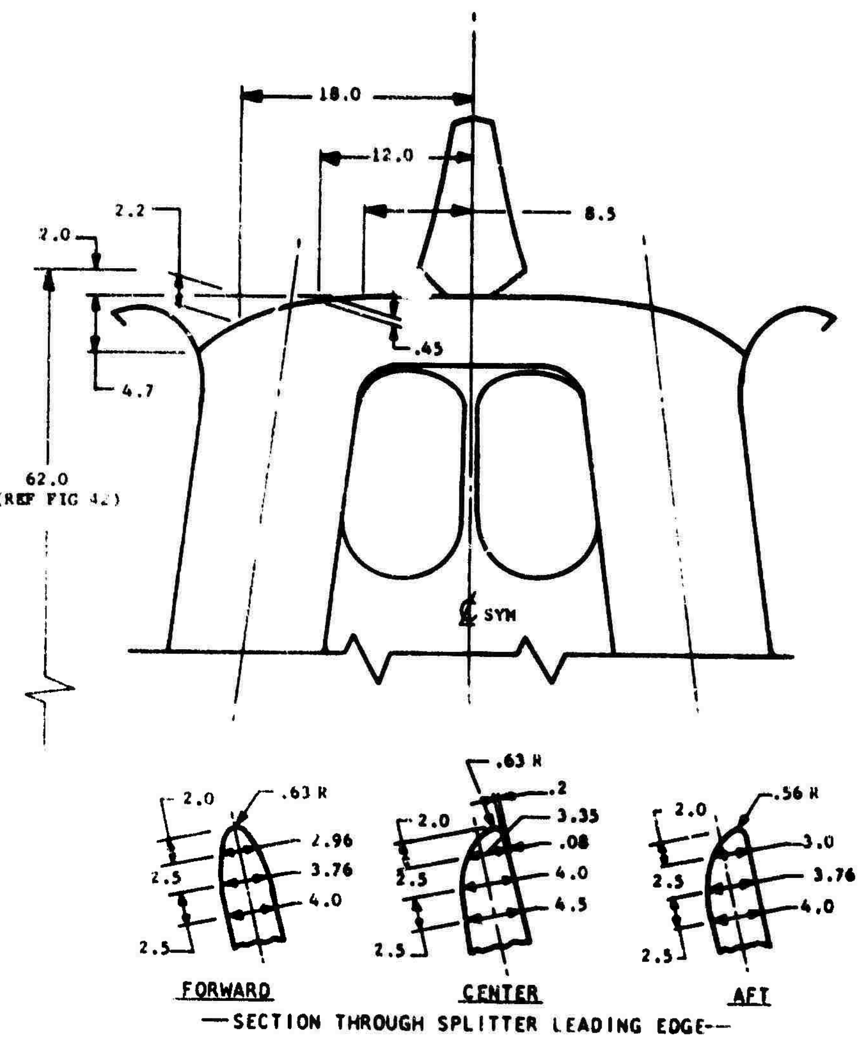


Figure 45. Ejector Splitter Leading-Edge Cross Sections for Configurations IV and V.

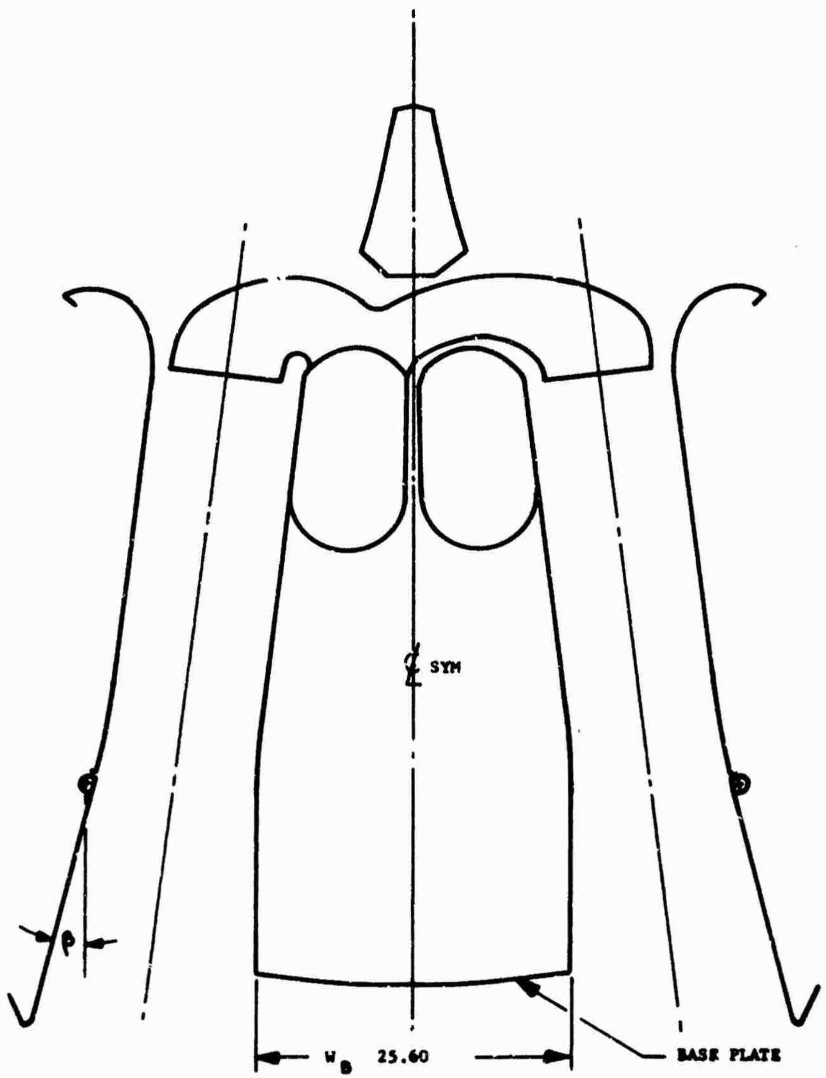


Figure 46. Ejector Cross Section for Configurations I and III.

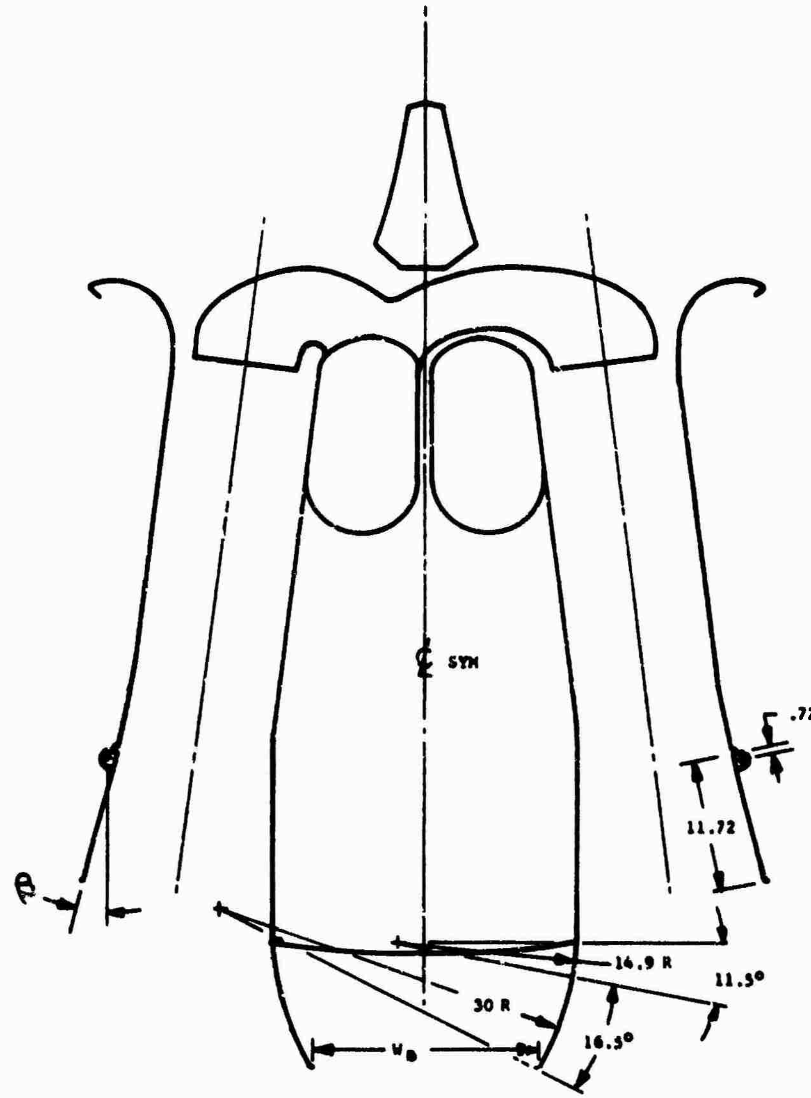


Figure 47. Ejector Cross Section for Configuration II.

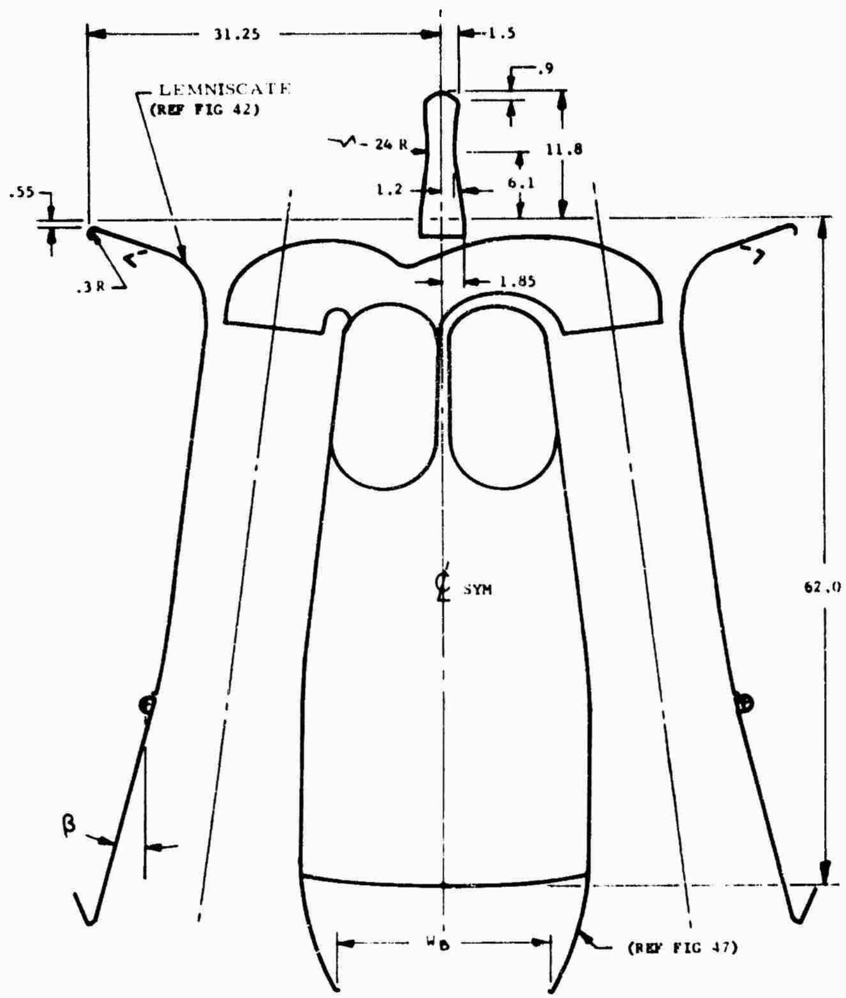


Figure 48. Ejector Cross Section for Configuration IV.

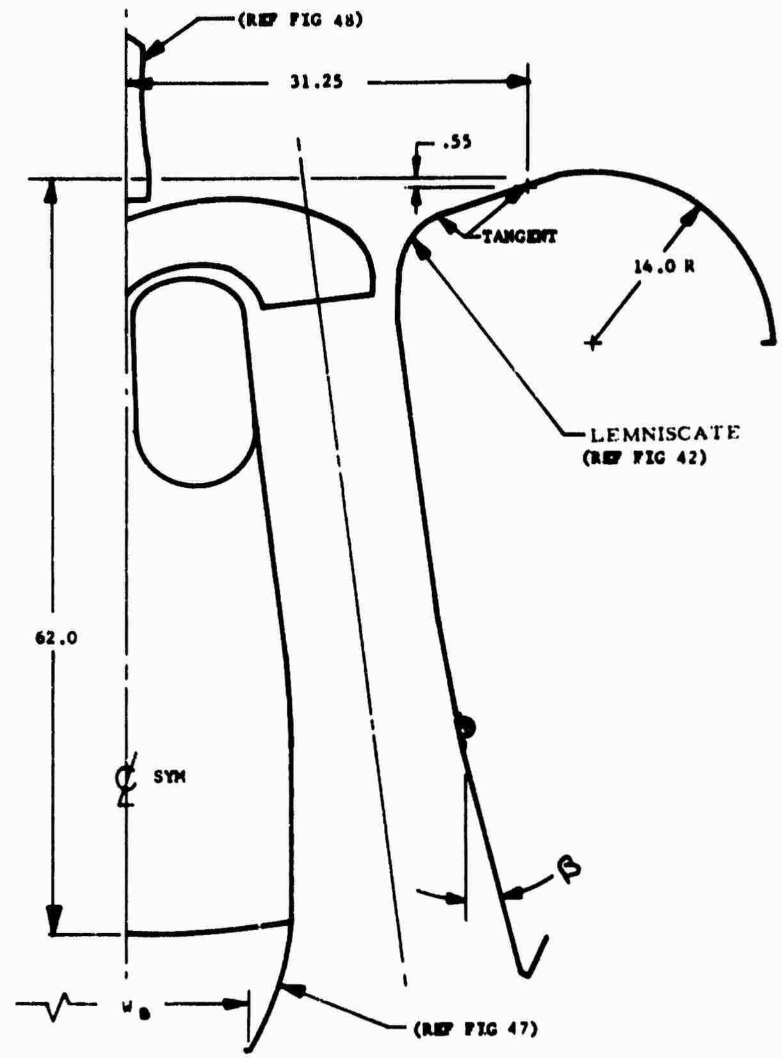


Figure 49. Ejector Cross Section for Configuration V.

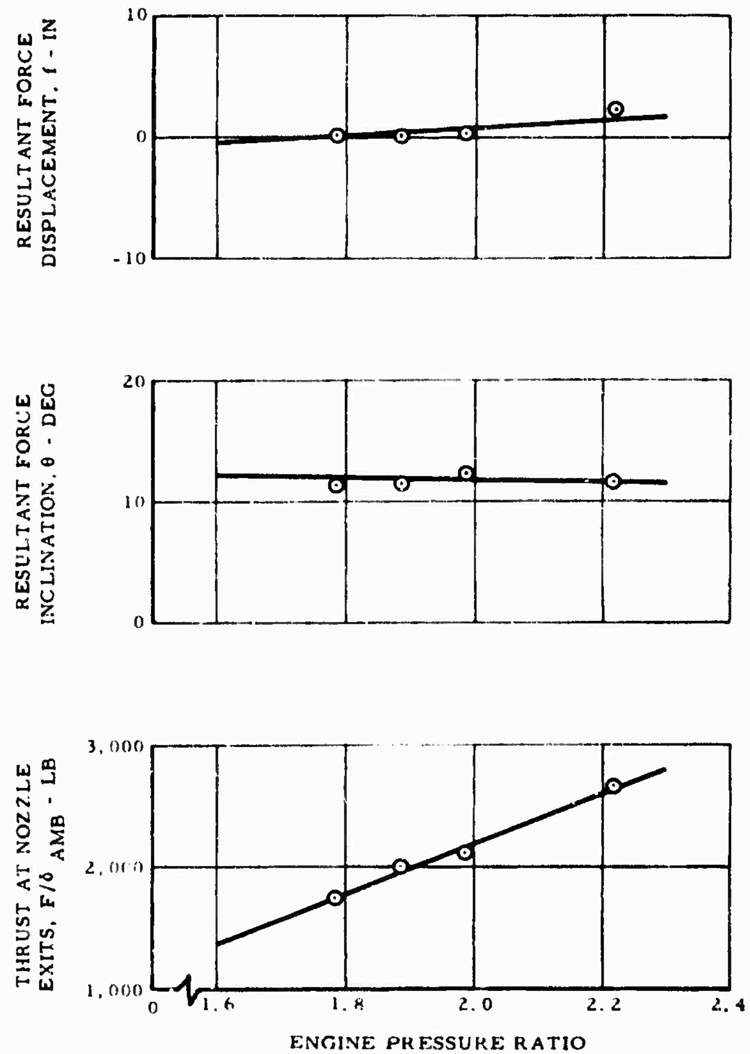


Figure 50. D Manifold Performance; Engine No. 1 Only.

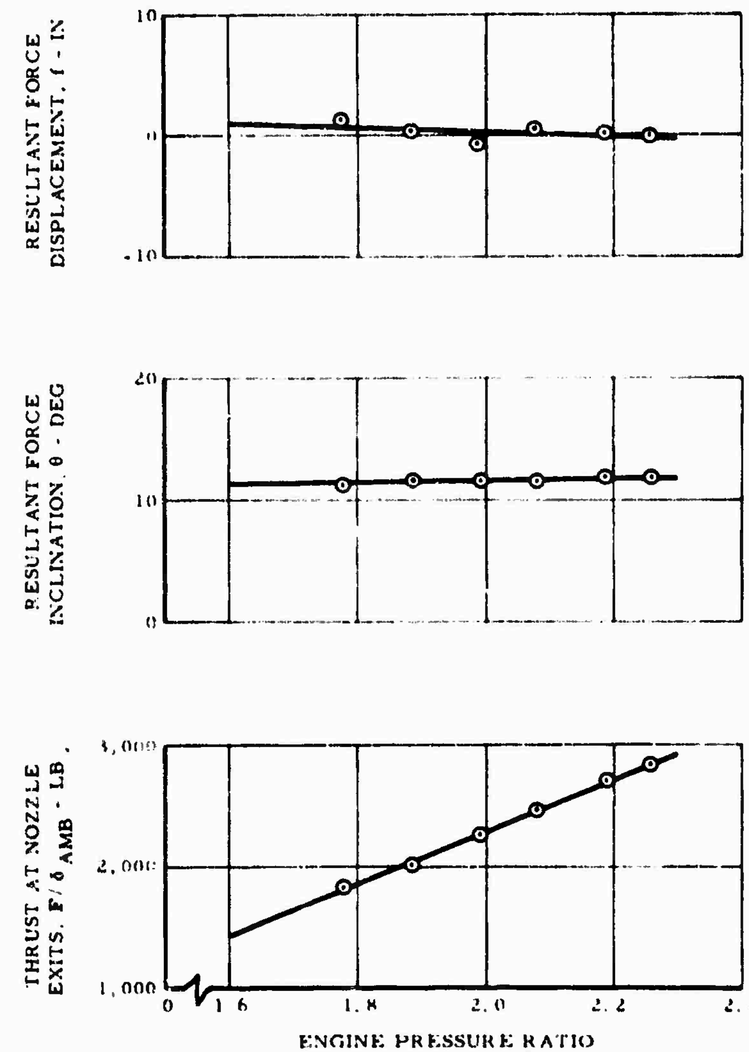


Figure 51. D Manifold Performance; Engine No. 2 Only.

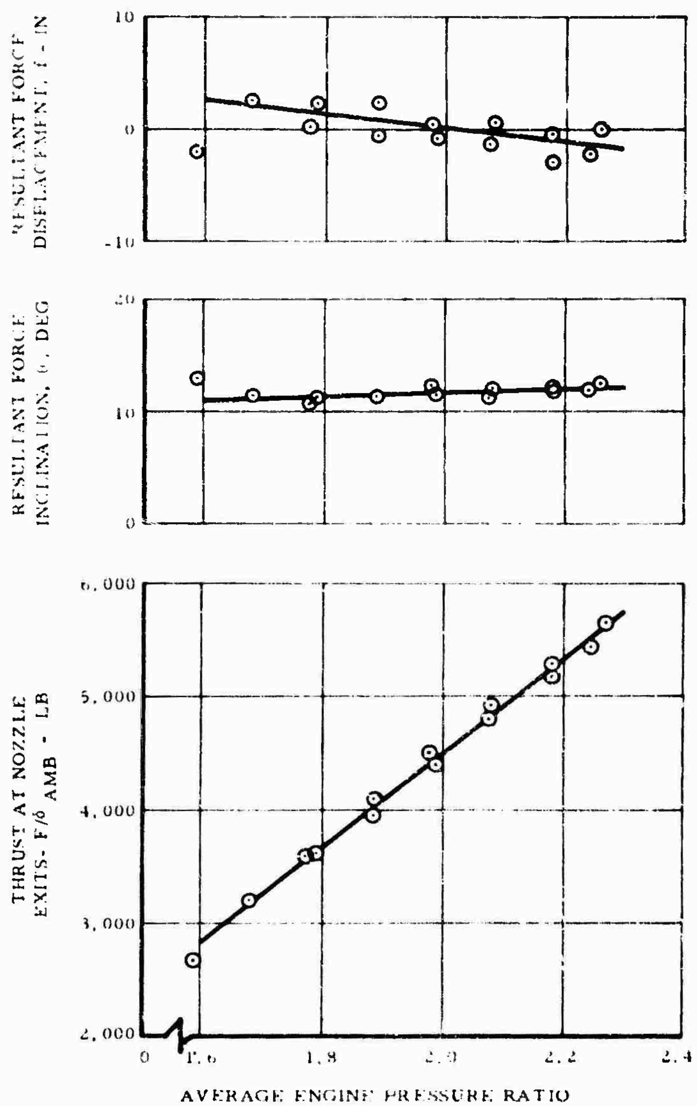


Figure 52. D Manifold Performance; Both Engines Operating.

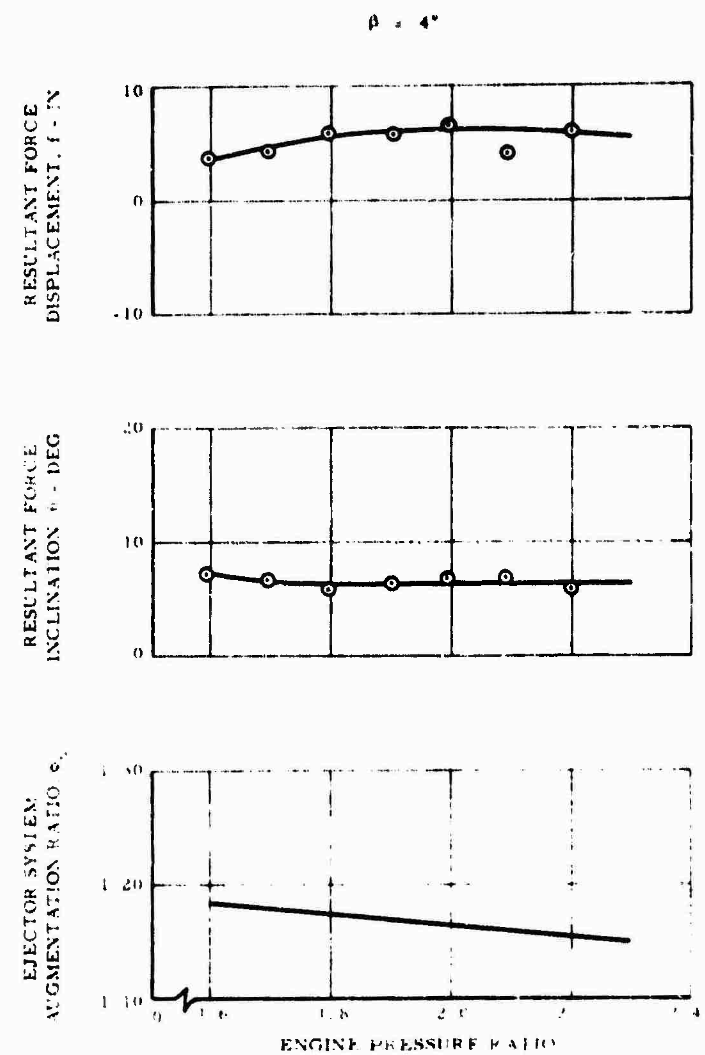


Figure 53. Ejector Configuration I Performance; Engine No. 1 Only.

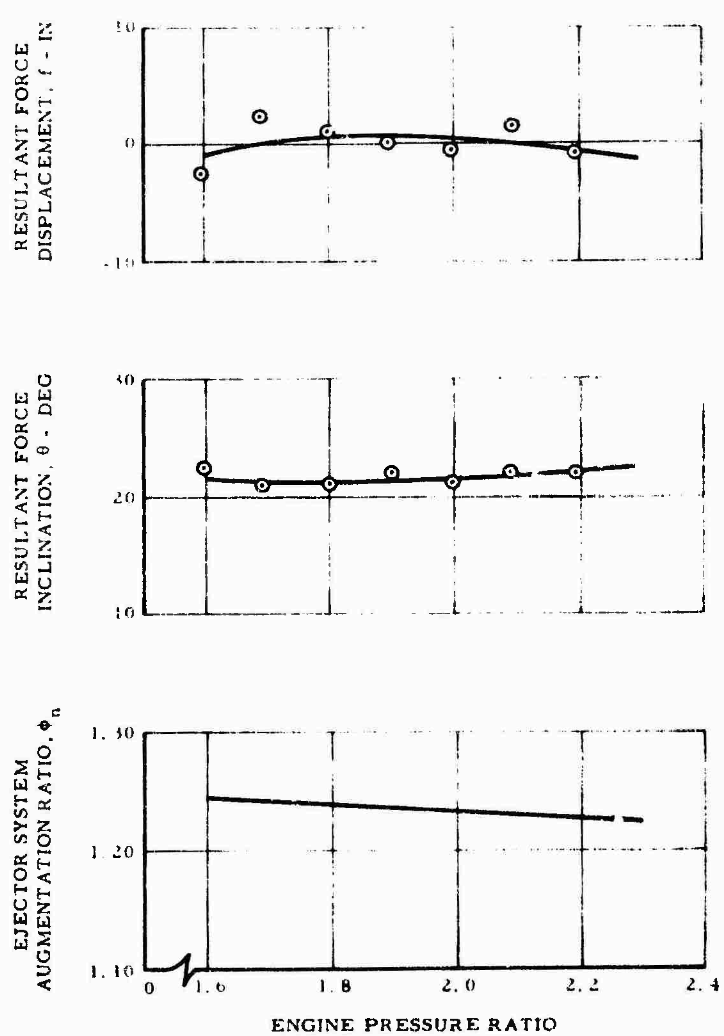


Figure 54. Ejector Configuration I Performance; Engine No. 2 Only.

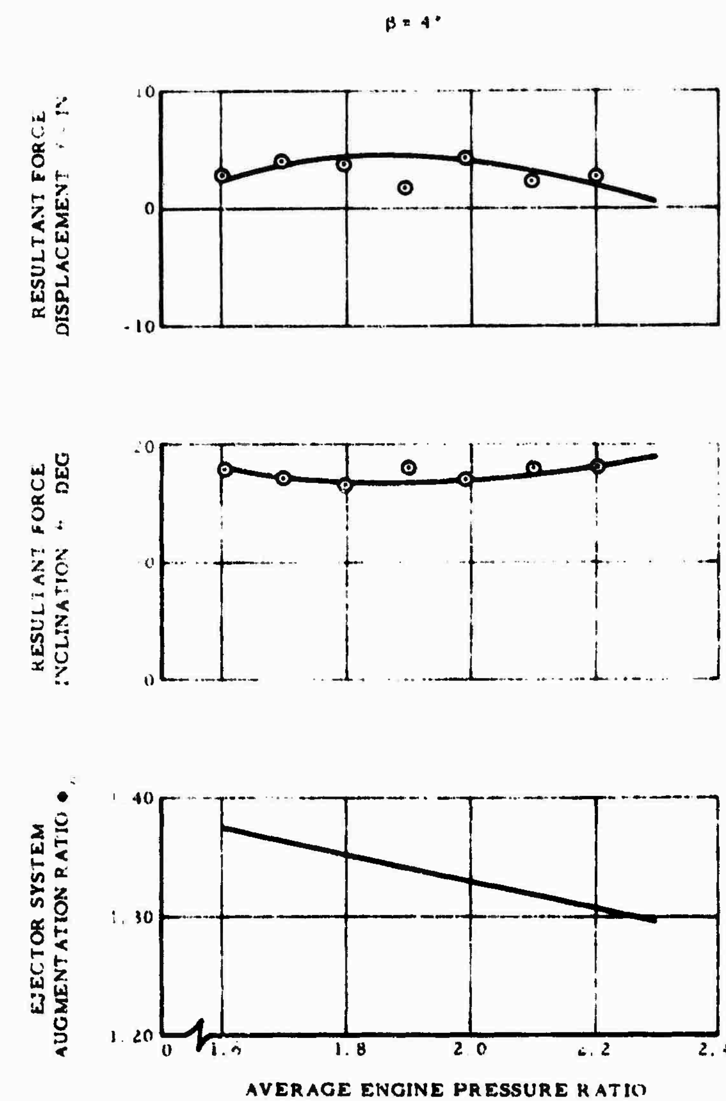


Figure 55. Ejector Configuration I Performance; Both Engines Operating.

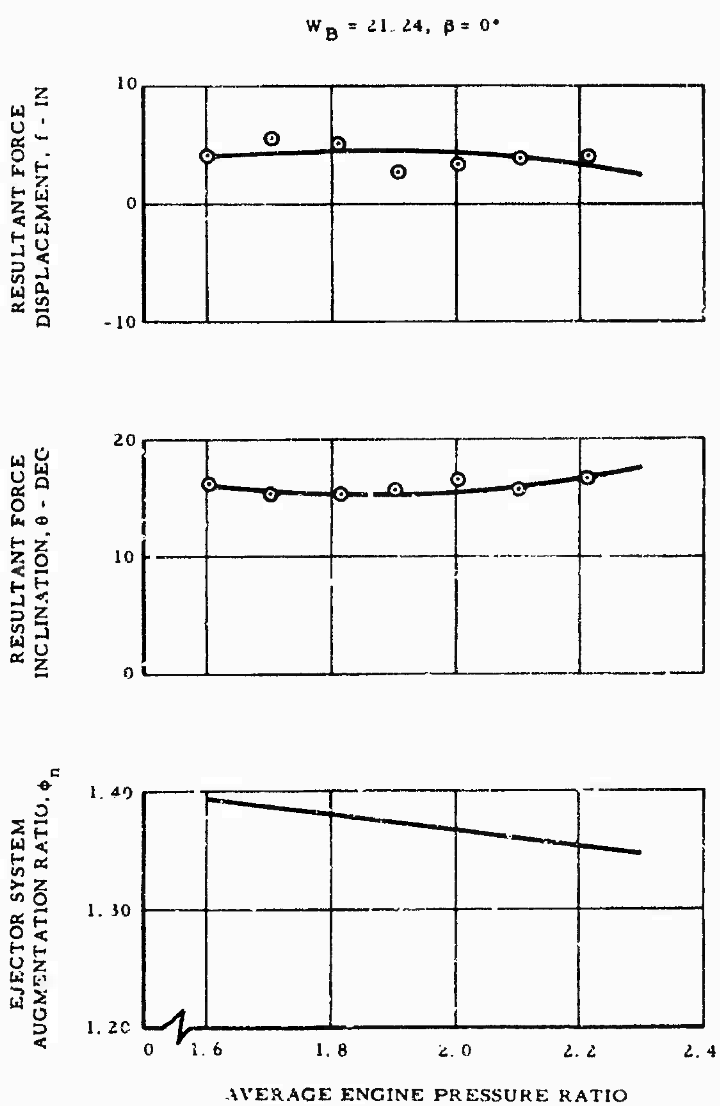


Figure 56. Ejector Configuration II Performance; Both Engines Operating.

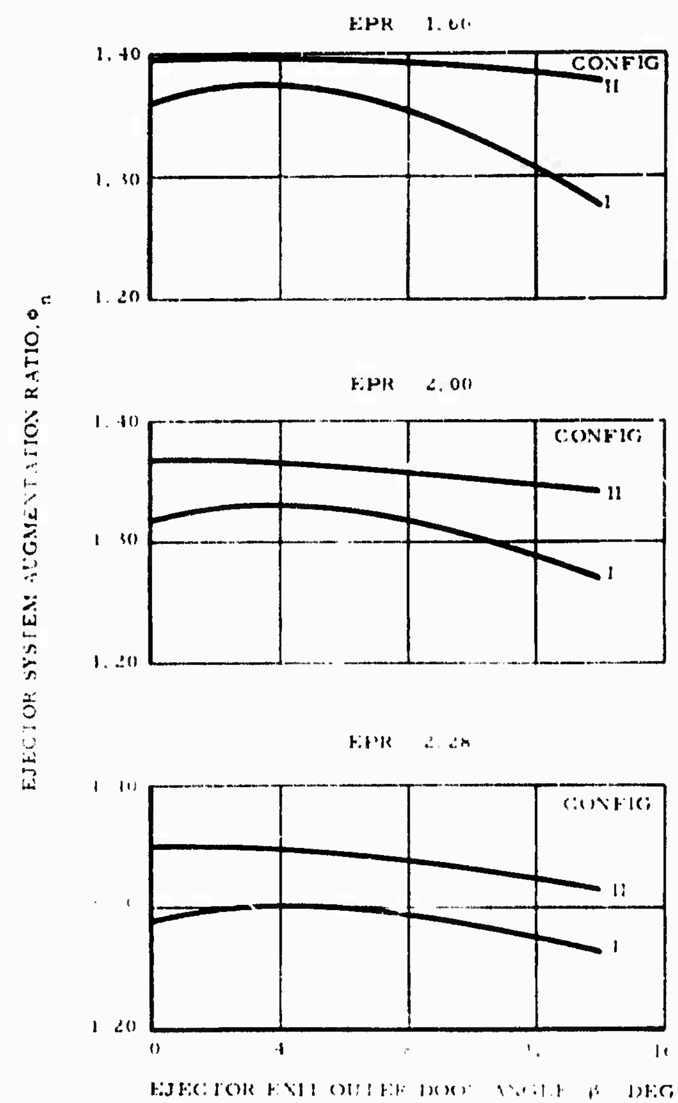


Figure 57. Comparison of Ejector Performance for Configurations I and II.

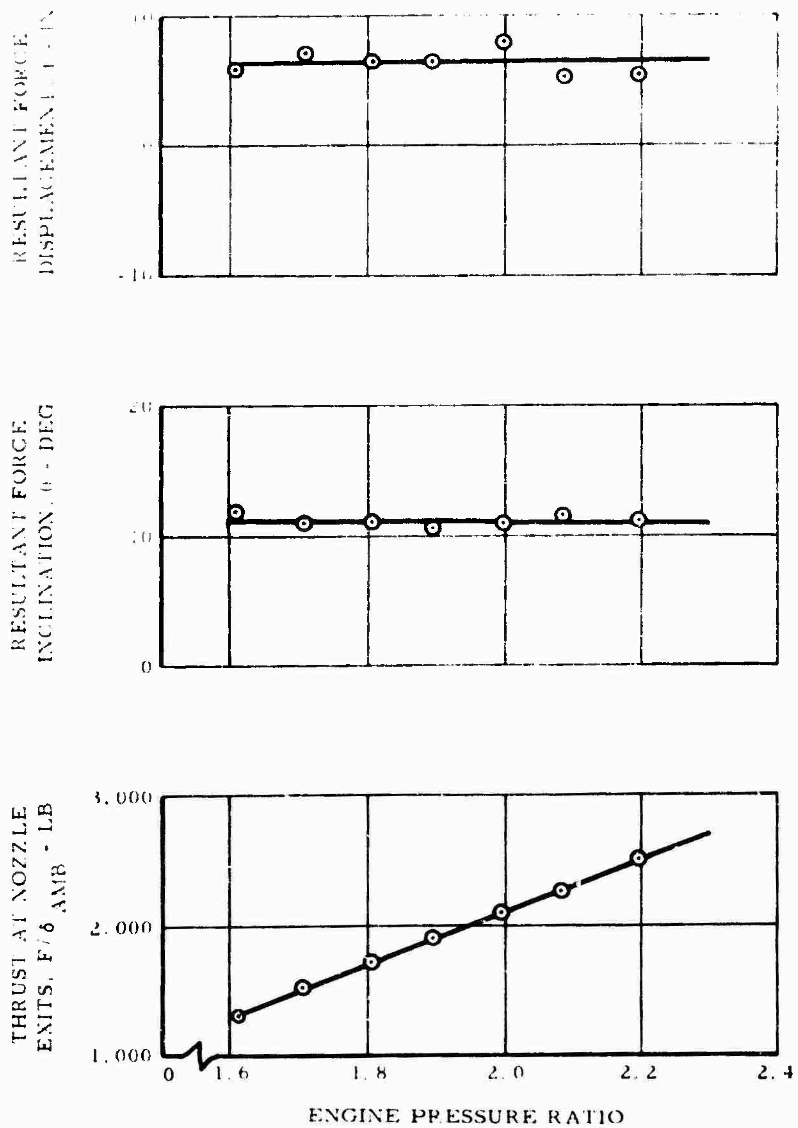


Figure 58. **F Manifold Performance; Engine No. 1 Only.**

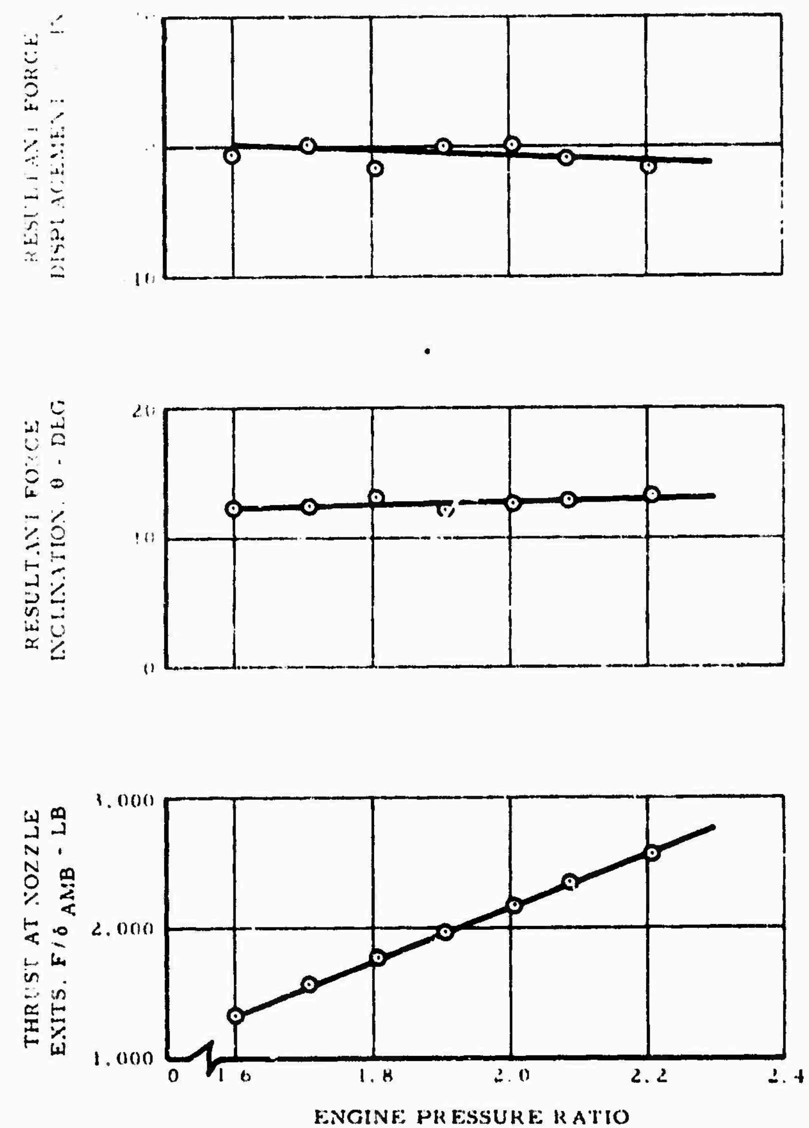


Figure 59. **F Manifold Performance; Engine No. 2 Only.**

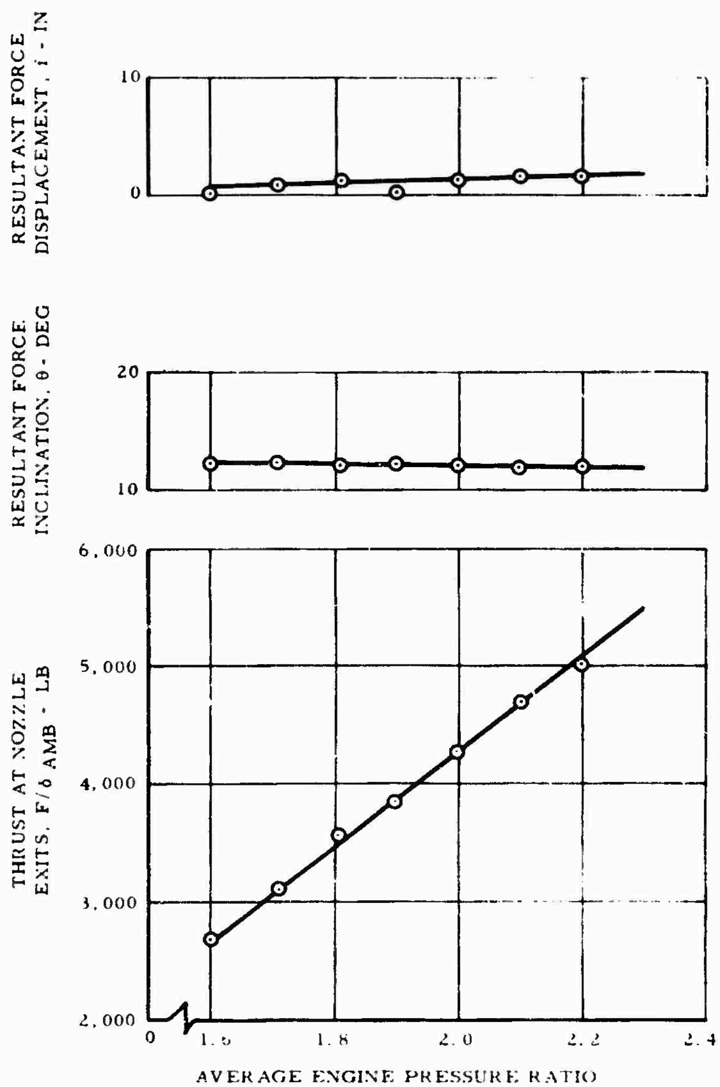


Figure 60. F Manifold Performance; Both Engines Operating.

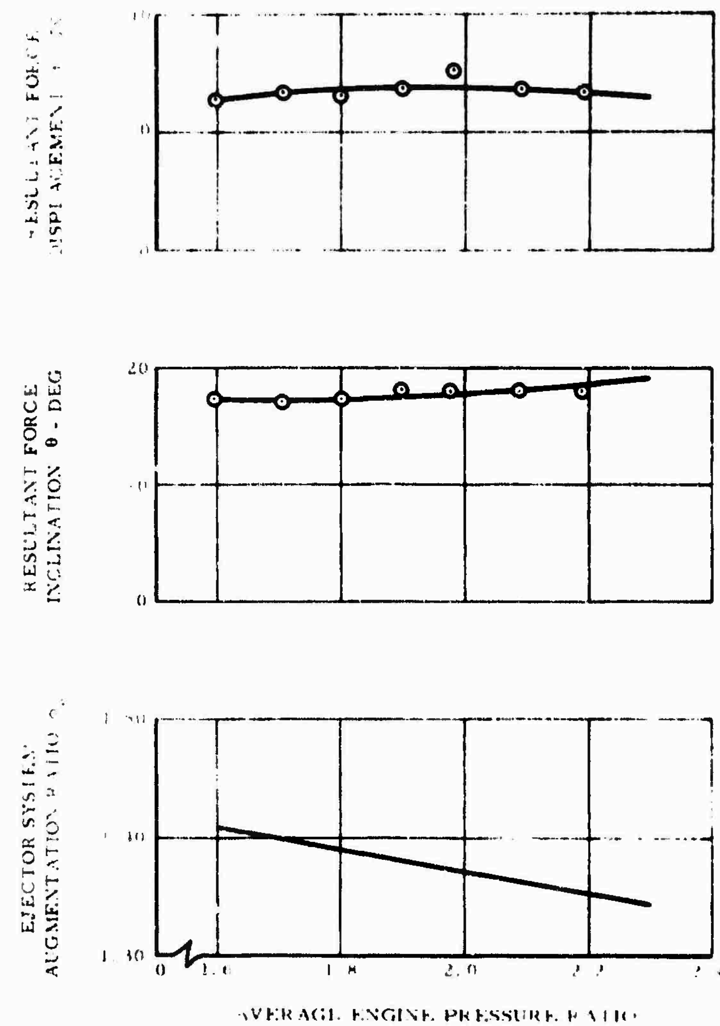


Figure 61. Ejector Configuration III Performance; Both Engines Operating.

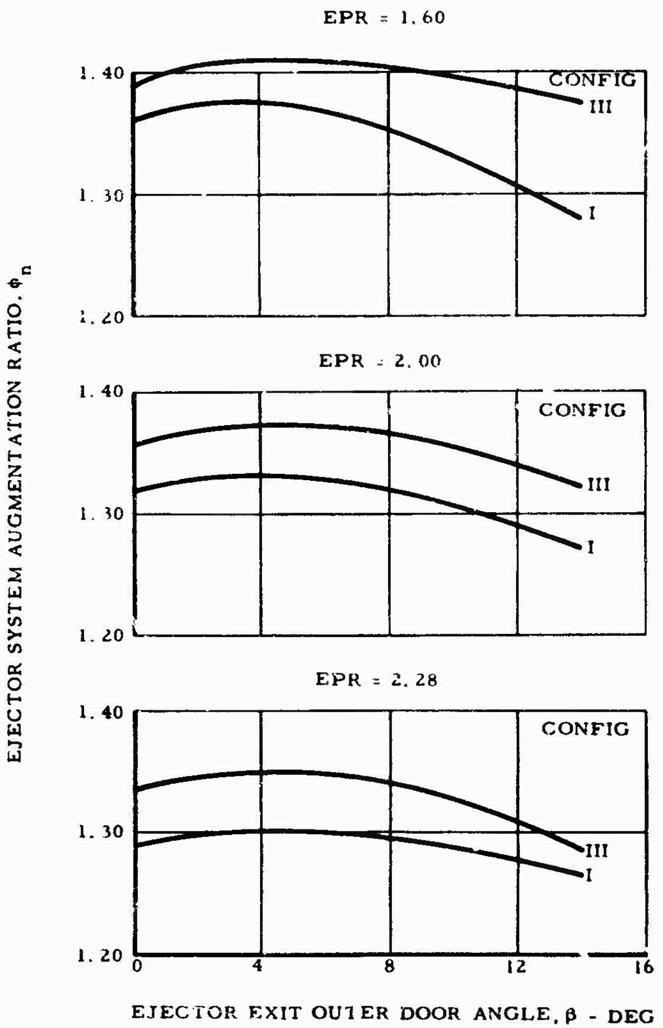


Figure 62. Comparison of Ejector Performance for Configurations I and III; Both Engines Operating.

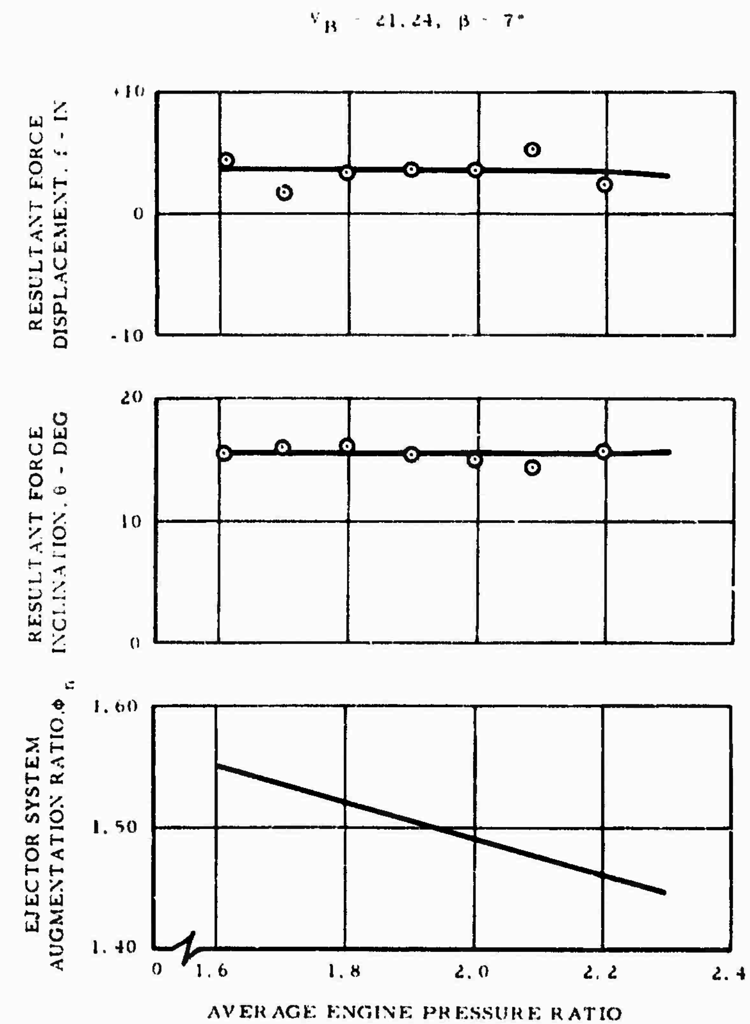


Figure 63. Ejector Configuration IV Performance; Both Engines Operating.

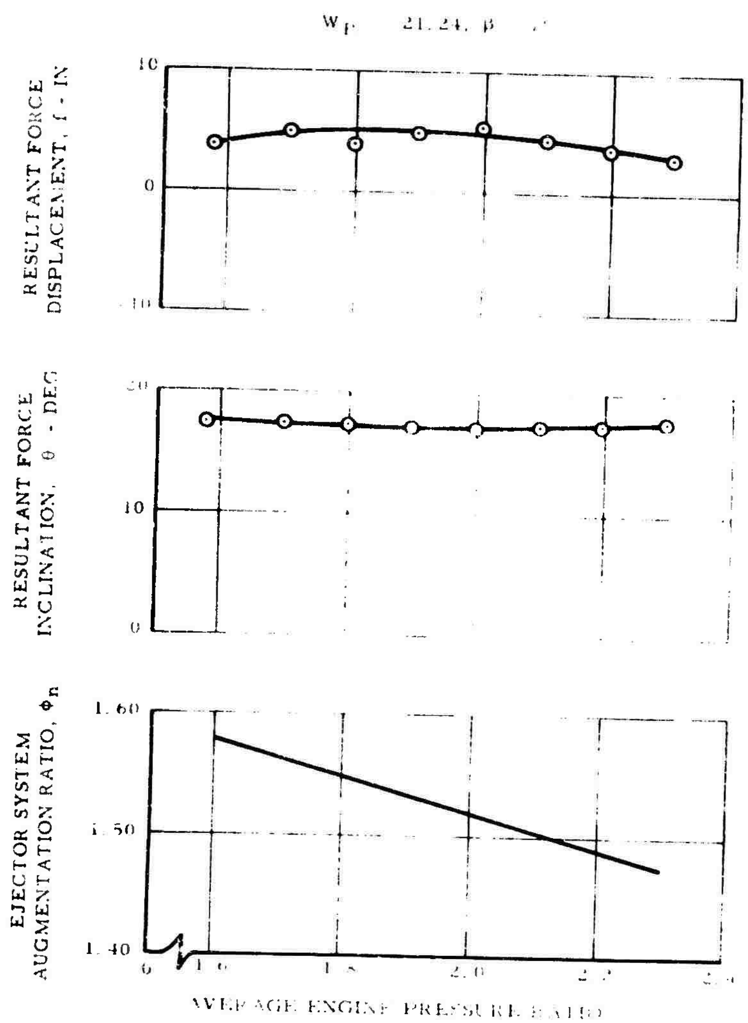


Figure 64. Ejector Configuration V Performance; Both Engines Operating.

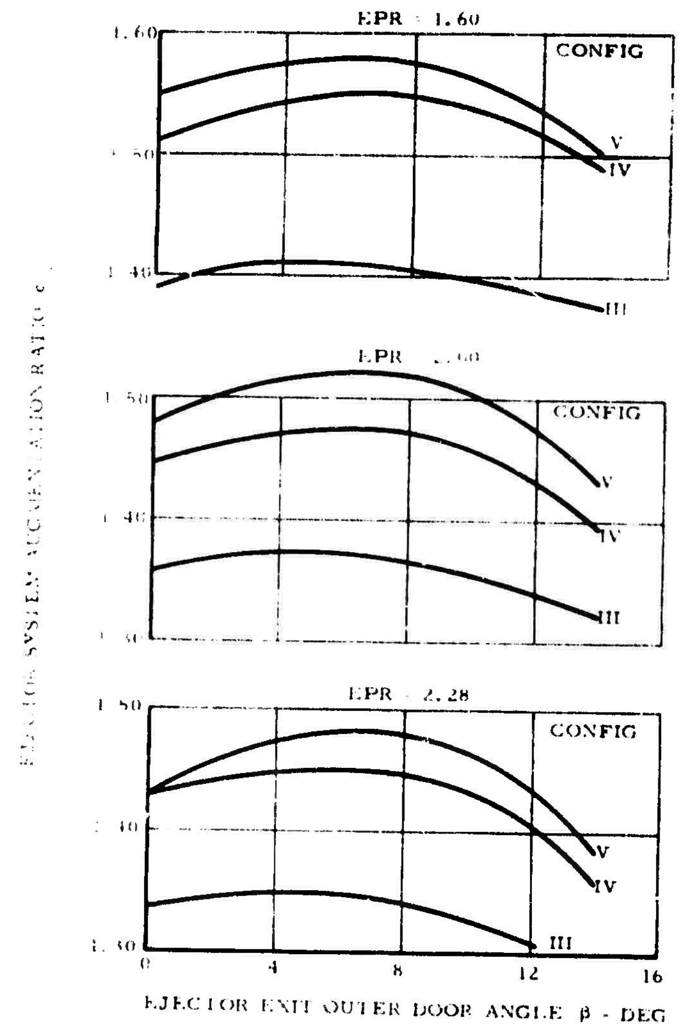


Figure 65. Comparison of Ejector Performance for Configurations III, IV, and V; Both Engines Operating.

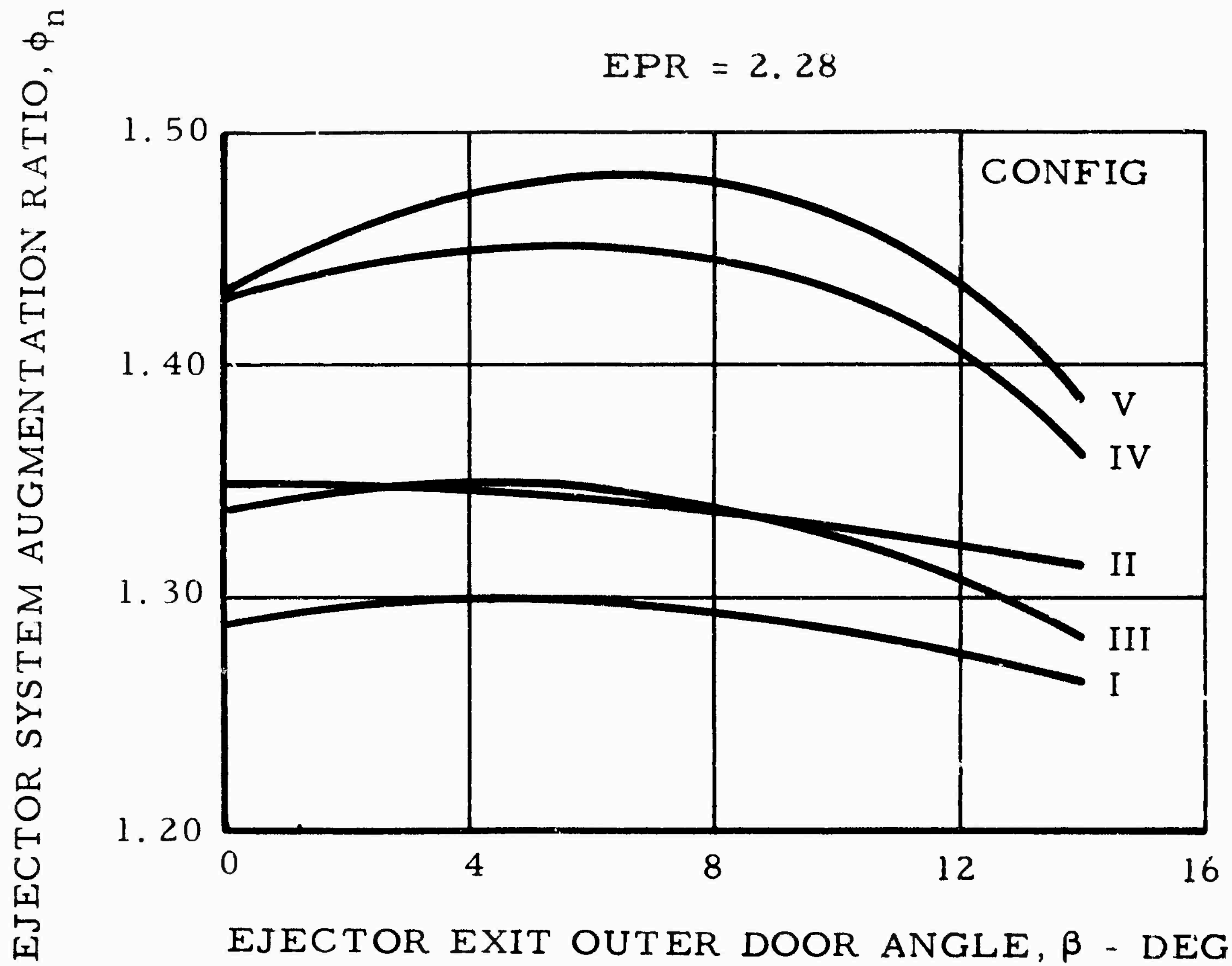


Figure 66. Comparison of Ejector Performance for Configurations I, II, III, IV, and V; Both Engines Operating.

PERFORMANCE FOR NO. 2 ENGINE AT AN EPR OF 2.20

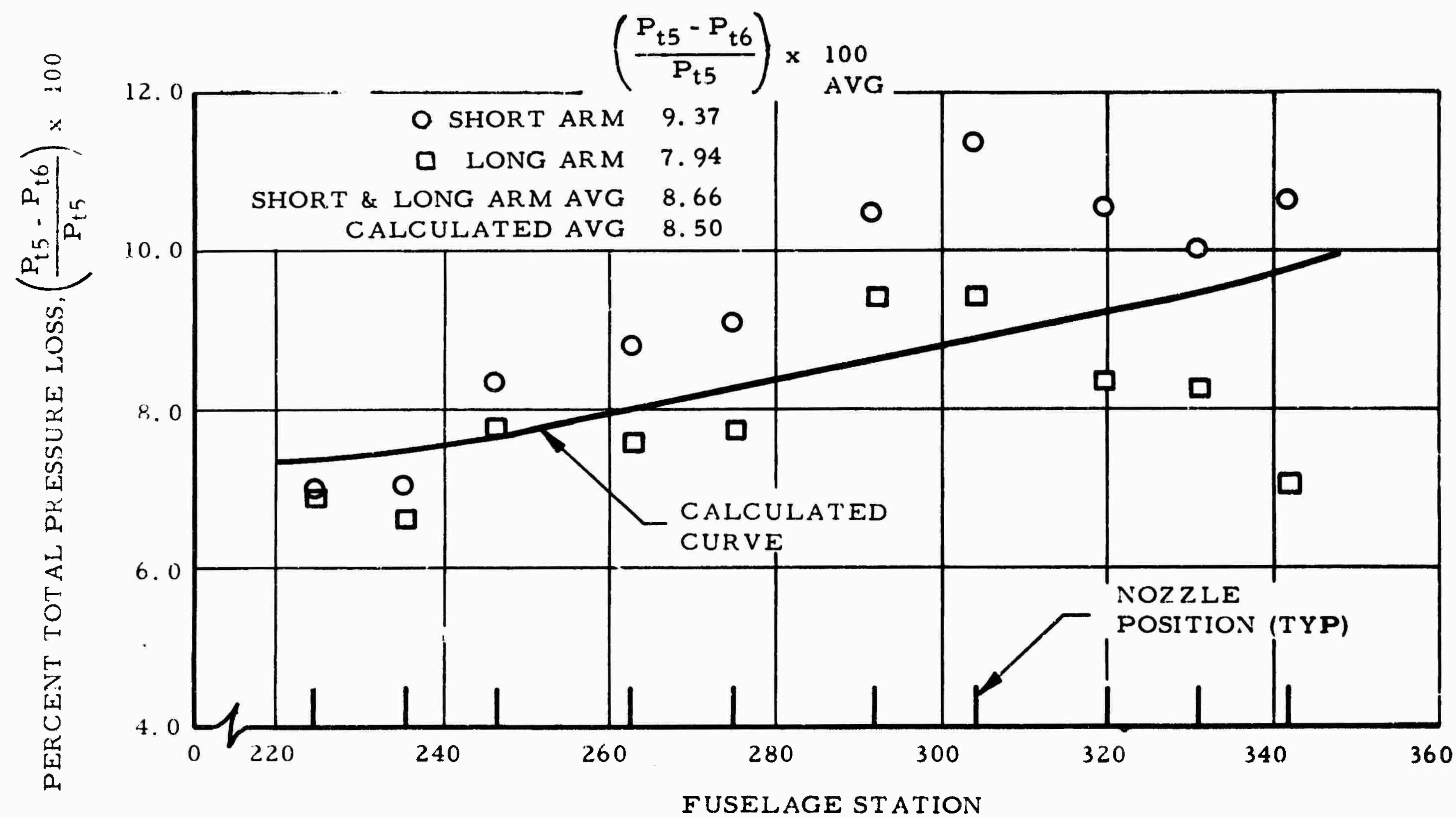


Figure 67. Engine Exit to F Manifold Nozzle Exit Total Pressure Loss Performance.

PERFORMANCE FOR NO. 2 ENGINE AT AN EGT OF 925°F

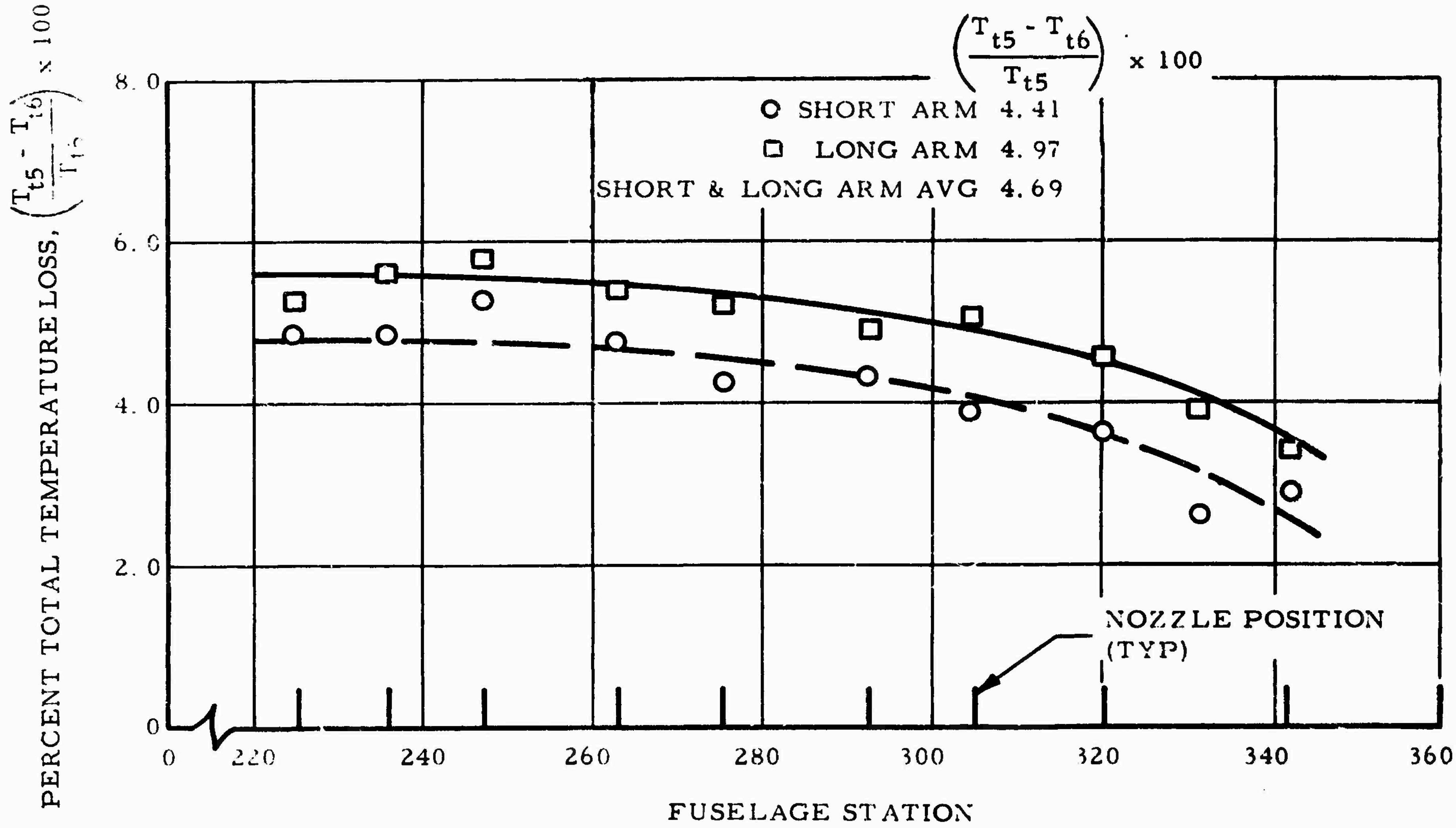


Figure 68. Engine Exit to F Manifold Nozzle Exit Total Temperature Loss Performance.

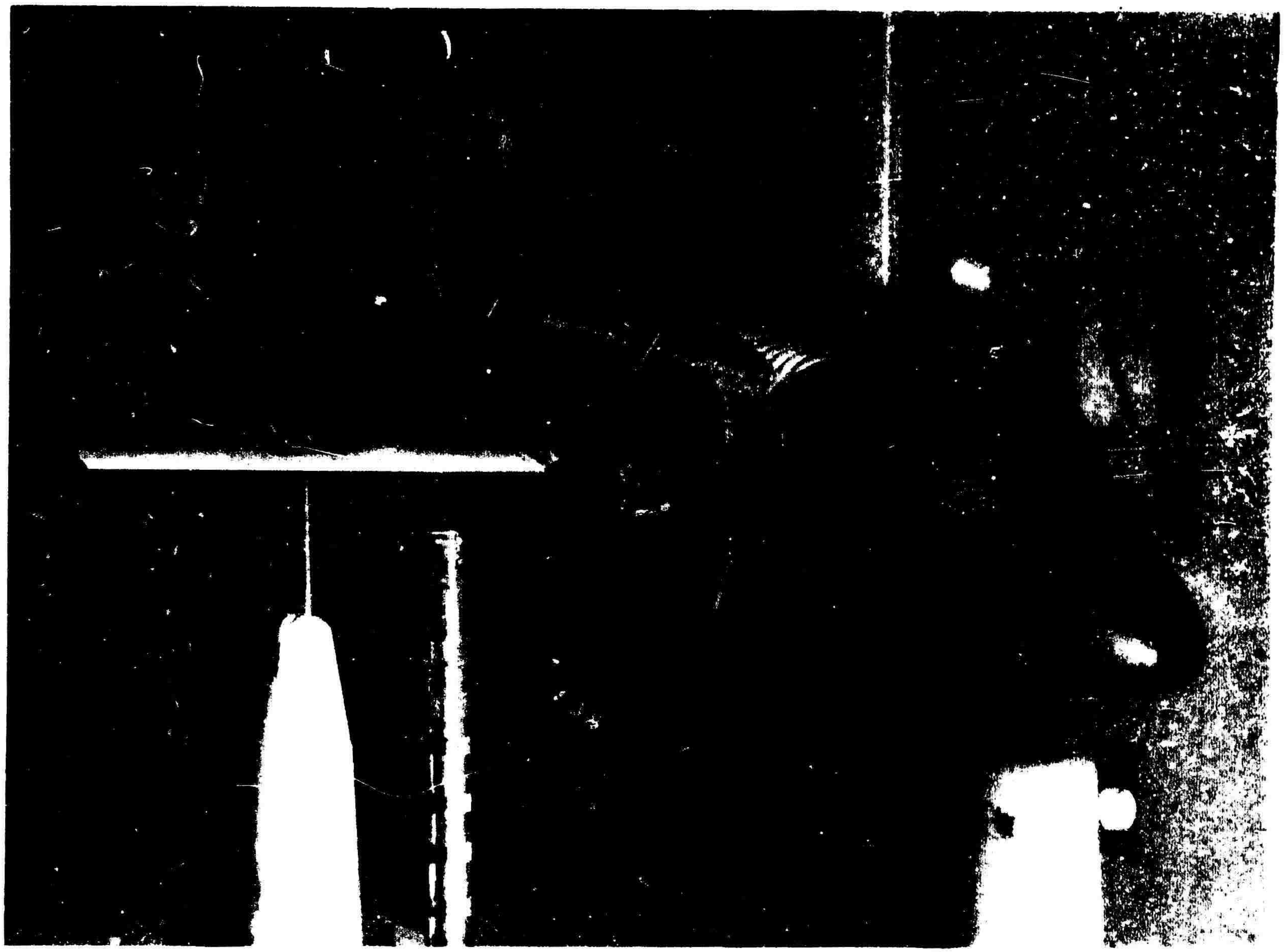


Figure 69. Small-Scale Wind Tunnel Model Mounted in Wind T..nnel.

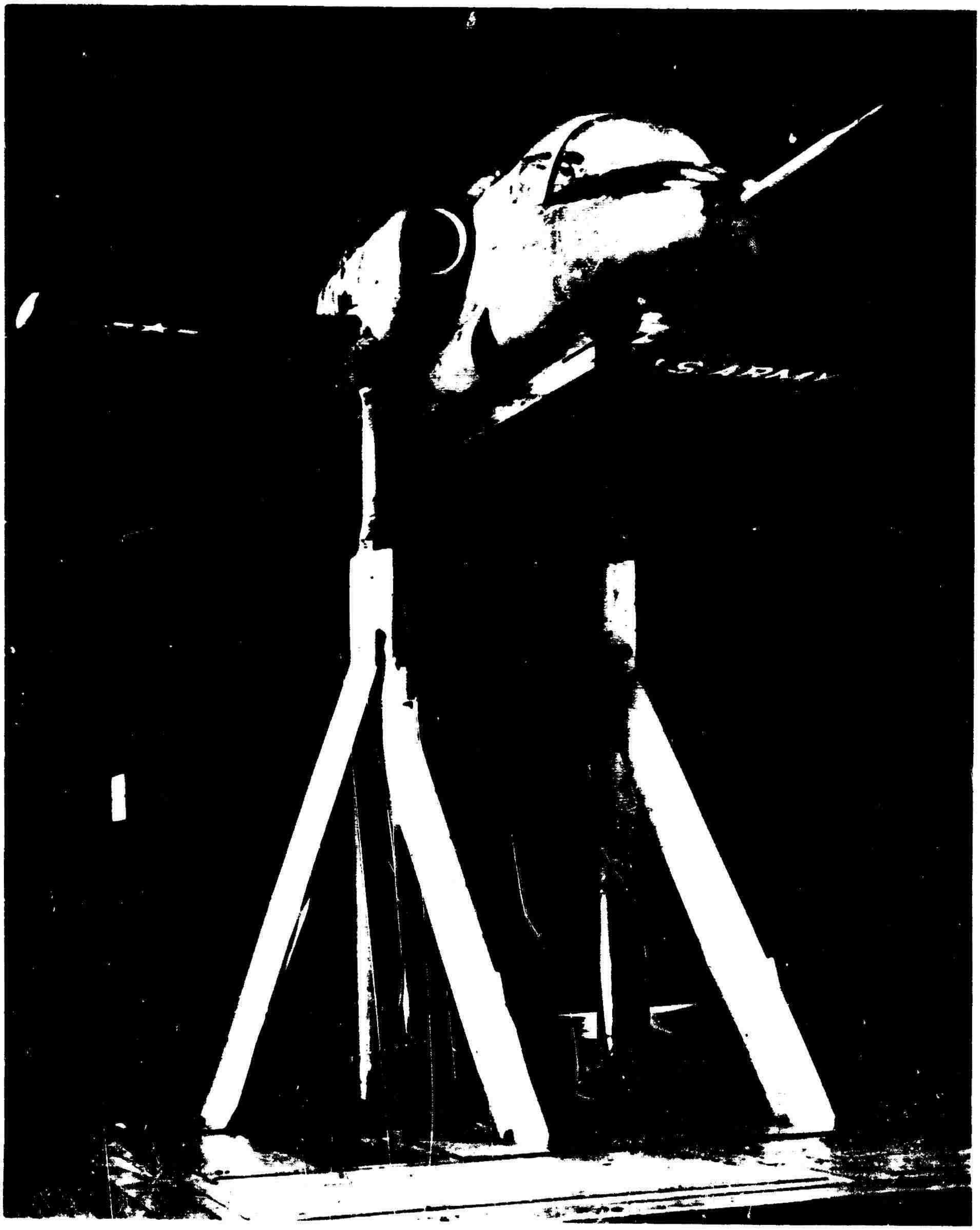


Figure 70. XV-4A Mounted in 40-by-80-Foot Wind Tunnel.

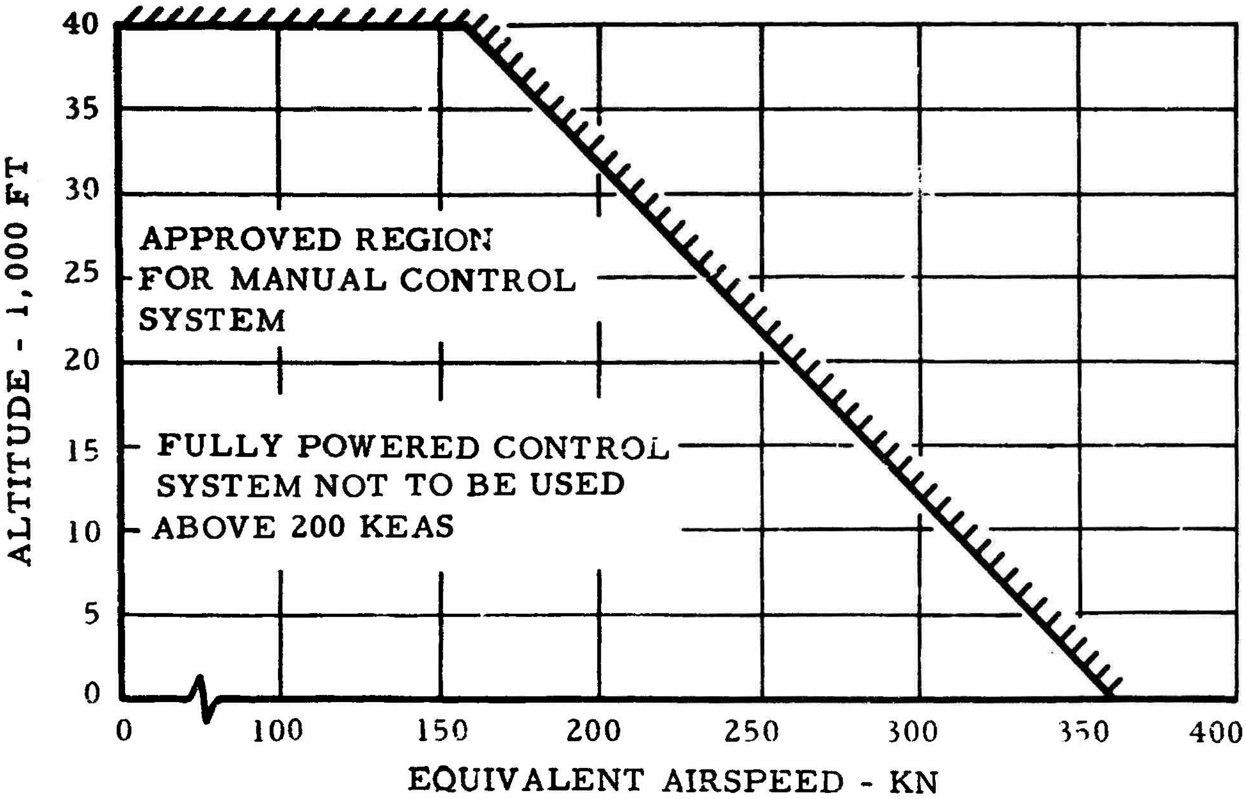


Figure 71. Flutter Airspeed Limits.

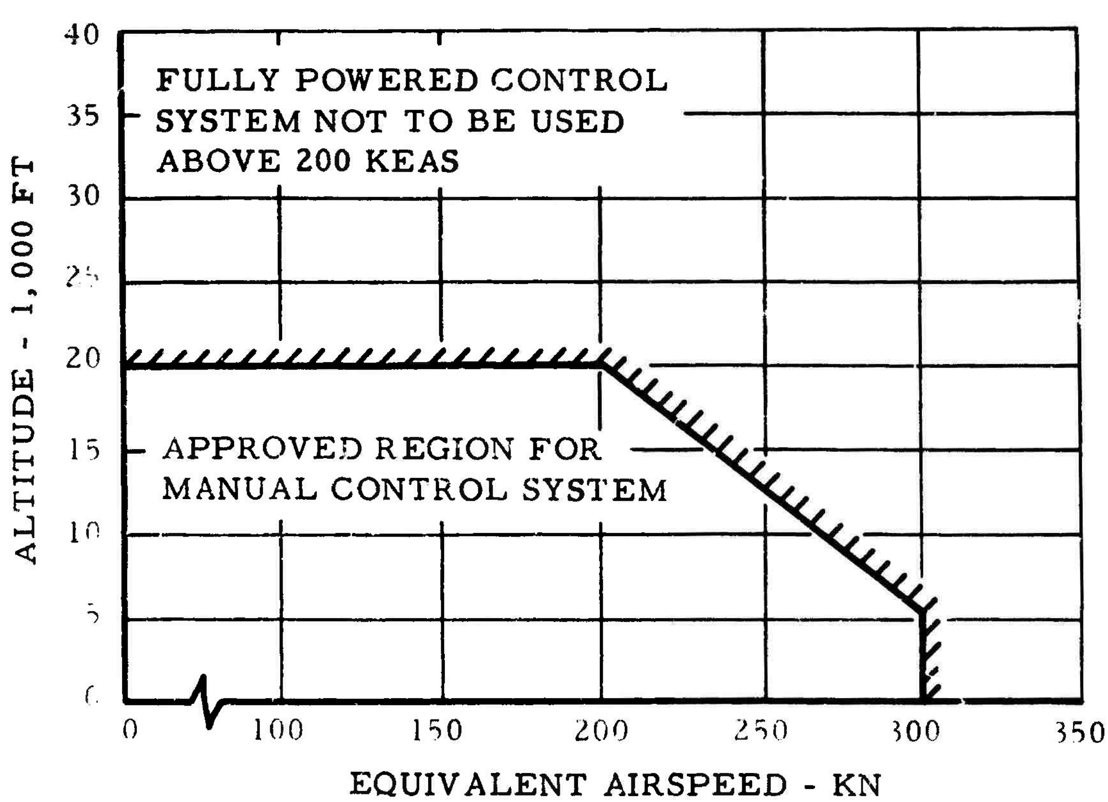


Figure 72. Placard Airspeeds.

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13. ABSTRACT

The results of the XV-4A VTOL research aircraft program, including a review of the aircraft design, aircraft systems, flight test program, VTOL lift improvement program, and small-scale and full-scale wind tunnel tests, are presented in this report.

DD FORM 1 JAN 64 1473

Unclassified

Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
XV-4A VTOL Aircraft Augmented Jet Ejector						

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